The ACEE Program and Basic Composites Research at Langley Research Center (1975 to 1986)

Summary and Bibliography

Marvin B. Dow

NASA

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Marvin B. Dow

Langley Research Center
Hampton, Virginia

Contents

Abstract

Composites research conducted at the Langley Research Center during the period from 1975 to 1986 is described, and an annotated bibliography of over 600 documents (with their abstracts) is presented. The research includes Langley basic technology and the composite primary structures element of the NASA Aircraft Energy Efficiency (ACEE) Program. The basic technology documents cited in the bibliography are grouped according to the research activity such as design and analysis, fatigue and fracture, and damage tolerance. The ACEE documents cover development of composite structures for transport aircraft.

Introduction

A crossroads event in the history of composites research at the Langley Research Center occurred in 1975. Over a span of years prior to 1975, the development of composites had proceeded in an orderly manner from laboratory-scale experiments to limited applications. As described in reference 1, the Langley research was focused in accordance with a 1972 Air Force–NASA Long-Range Planning Study for Composites (RECAST). The event that would cause a fundamental change was the formation of the Aircraft Energy Efficiency (ACEE) Program.

From 1976 until its termination in 1985, the ACEE Program was the central element in NASA composites research. The rationale, planning, and implementation of the program are discussed in references 2 and 3. Composite structures were one element of a comprehensive plan for developing aeronautical fuel-conservation technology. The goal of the Composite Primary Structures element of the ACEE Program was to accelerate the application of composites to primary structures in new civil transport aircraft by (1) development of design and manufacturing techniques for composite empennage, wing, and fuselage structures, (2) dissemination of technology throughout the transport industry, and (3) extensive flight service evaluations. ACEE composites research was performed under contracts with the transport builders and managed by a small project staff at Langley.

During the ACEE era, Langley personnel provided expert assistance to the ACEE Project Office but primarily conducted a program of traditional, or base, research. This research was performed at Langley or under numerous grants and contracts with Langley and covered the aerospace spectrum from helicopters to airplanes to spacecraft. Together the ACEE and Langley base research programs produced results of major significance to composites technology in the United States.
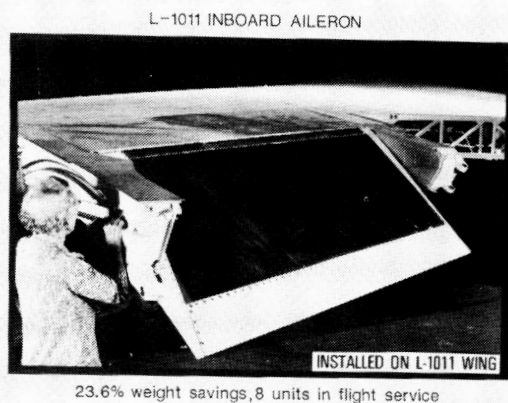
This report briefly summarizes the ACEE and base programs in composite materials and structures and presents an annotated bibliography of the many reports and documents on the subject produced during the 11-year period from 1975 to 1986. The bibliography of over 600 publications (with their abstracts) is organized into subsections according to research disciplines or aircraft structural components. An author index to these documents is also provided. This report deals only with resin-matrix composite materials and with research conducted under the auspices of the Langley Research Center. A similar summary including a bibliography covering ACEE aerodynamics research is presented in reference 4.

ACEE Composites Program

Composite primary structures became an element in the ACEE Program because composites offered a means to conserve fuel use by transport aircraft. Studies such as that reported in reference 2 indicated that extensive use of composites in major structural components could reduce aircraft structural weight by 25 percent or more and, as a consequence, save 10 to 15 percent in fuel usage. The planned application of composites would require the development of revolutionary technology in aircraft structures. Moreover, extensive use of composites would require the following barriers to be overcome:

1. Experience with composites resided with research groups rather than with designers and manufacturers.
2. Uncertainties in the development and production costs of composites made it difficult for them to compete with established aluminum technology.
3. Long duration performance and maintenance requirements of composites were unknown, so that users were reluctant to accept this new material.

A systematic building block approach was selected to achieve the ultimate goal of composite wings and fuselage on transport aircraft. Development began with lightly loaded secondary structural components, proceeded to medium primary structural components, and was planned to conclude with wings and fuselage. Although the ACEE composites program was terminated before completion of wing and fuselage development, important results were obtained. This section summarizes the component developments and technology research accomplished during the ACEE era. Details are contained in the reports

DC-10 UPPER AFT RUDDERS

26.4% weight savings, 12 units in flight service

L-1011 INBOARD AILERON

23.6% weight savings, 8 units in flight service

B-727 ELEVATORS

24.6% weight savings, 10 units in flight service

DC-10 VERTICAL STABILIZER

7 ft x 23 ft, 780 lb, 22.6% weight savings

L-1011 VERTICAL FIN

9 ft x 25 ft, 620 lb, 28.4% weight savings

B-737 HORIZONTAL STABILIZER

4 ft x 17 ft, 204 lb, 22.1% weight savings

(a) Secondary components.

(b) Medium primary components.

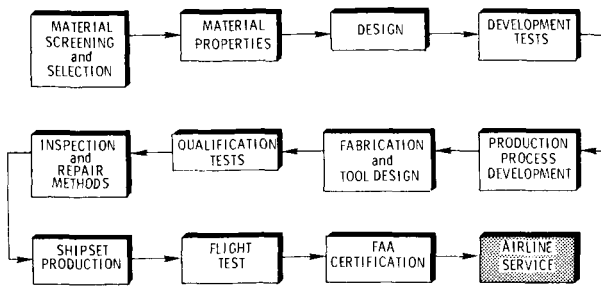Figure 1. Composite transport structural components developed under ACEE Program.

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Figure 2. Elements of typical component development contract.

Figure 3. Graphite-epoxy upper aft rudder on DC-10.

Figure 4. Graphite-epoxy elevator on B-727.

cited in the annotated bibliography under "ACEE Composites Program" and "Conference Documents."

Transport Structures Development

Under the ACEE Program, Boeing Commercial Airplane Co., Douglas Aircraft Co., and Lockheed Corp. contracted to develop the secondary and empennage (medium primary) components shown in figure 1. The weight savings indicated for each component is based on the weight of the original aluminum alloy component. Although different in detail, each contract encompassed the elements shown in figure 2 and each incorporated cost-sharing provisions. In addition to achieving technical goals, each development was to acquire those cost data required for the builders to make production commitments. A common element in the components was the use of Narmco T300/5208 (graphite-epoxy), a graphite-fiber-reinforced thermoset matrix material cured at 350°F. Each contractor elected to use company funds to acquire the design allowables data required for Federal Aviation Administration (FAA) certification. Only the Lockheed data (ref. 5) were published.

Douglas DC-10 upper aft rudder. Development of the upper aft rudder, a secondary component, on the DC-10 began in 1974 under the Langley base program and was completed under the ACEE Program. Structural arrangement of the rudder is shown in figure 3, and details of the design, testing, and qualification are given in reference 6. Two noteworthy features of the rudder development (ref. 7) were the postbuckled (tension field) design and the "trapped rubber" manufacturing process. Twenty composite rudders were manufactured for flight service, which began in June 1976. Except for isolated damage incidents, primarily due to lightning strikes, flight service of the rudders has been uneventful. Flight service data are reported with similar data from other components in Langley reports. (See the section in the bibliography under the base program on "Environmental Effects and Flight Service.")
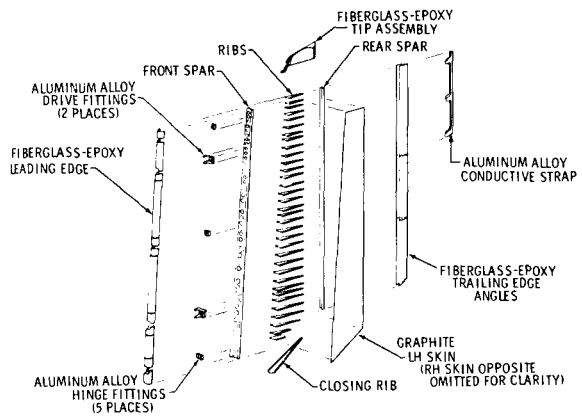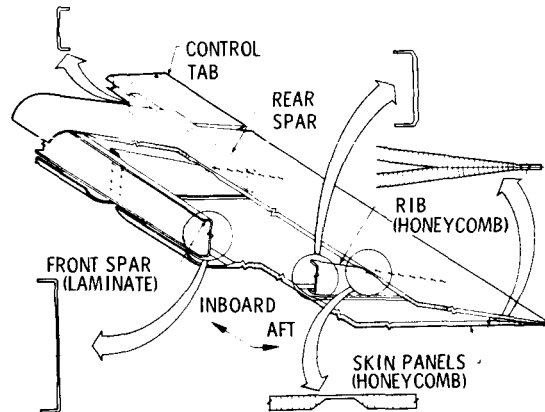
Boeing B-727 elevator. The structural arrangement of the composite elevator on the B-727 is shown in figure 4; additional details are available in reference 8. Boeing selected a design featuring a honeycomb-stiffened skin and a conventional manufacturing process in which individual elements were autoclave cured and then mechanically assembled. Five shipsets (10 elevators) were manufactured for flight service, which began in March 1980. As for the DC-10 rudder, service data are reported in Langley reports listed under "Environmental Effects and Flight Service" in the bibliography. Boeing credits successful development of the elevator with providing the confidence and experience needed to use composite components on the B-757 and B-767 transports.

Lockheed L-1011 inboard aileron. The aileron structure is located behind the wing engines on the L-1011 and has the structural arrangement shown in figure 5. Details of the composite aileron design, manufacture, and testing are given in

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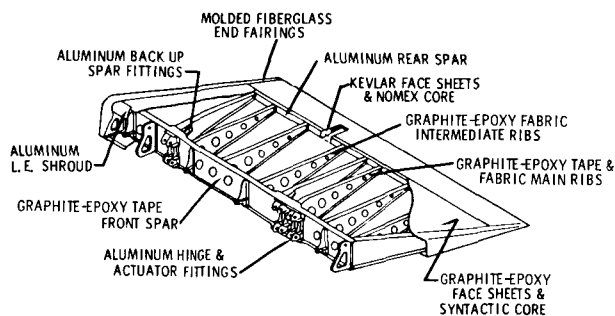
Figure 5. Graphite-epoxy inboard aileron on L-1011.

Figure 6. Graphite-epoxy vertical fin on L-1011.

(a) Spar segment.   (b) Cover segment.

Figure 7. Composite elements used for Production Readiness Verification Tests. (Linear dimensions are in inches.)

reference 9. Aileron manufacture and assembly were performed by AVCO Corporation under subcontract to Lockheed. This contractual arrangement was similar to that for the L-1011 metal wing structure. The composite aileron features innovative sandwich cover panels with cores constructed of epoxy syntactic foam. Five shipsets of ailerons were manufactured for flight service, which began in March 1982. Annual flight service summaries are published as NASA Contractor Reports, while cumulative data appear in Langley compilations.
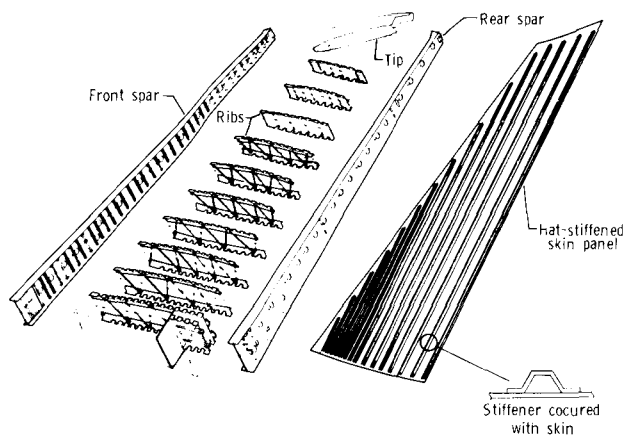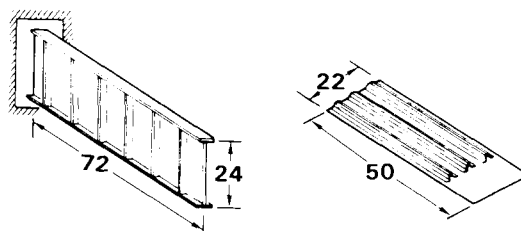
Lockheed L-1011 vertical fin. The L-1011 vertical fin was the first medium primary composite structure developed; the contract was awarded in 1975. Work began under auspices of the Langley base technology program and was later transferred to the ACEE Program. The development was a joint effort by Lockheed's California and Georgia Companies.

The fin, shown in figure 6, is a conventional two-spar structure with interspar ribs and stiffened-skin panels. The composite design incorporated hat-stiffened skin panels and C-section spars. Mechanical fasteners were used extensively in assembly of subelements. Reference 10 summarizes the design, manufacture, and testing of the composite fin. In addition to extensive development tests, Lockheed

conducted a study called Production Readiness Verification Tests. This study, reported in reference 11, provides valuable information on strength variations in composite elements (shown in fig. 7) before and after exposure to simulated flight environments.

After a lengthy and problem-plagued development, the fin program concluded with full-scale tests. The composite fin experienced a failure at less than design ultimate load during static testing, the failure resulting from unanticipated secondary loading effects. The failure and corrective action are discussed in reference 12, which also discusses failure events experienced by the Boeing and Douglas empennage components. Static and fatigue tests were successfully completed on a second test article. The full-scale tests were performed, documented, and witnessed in accordance with FAA certification requirements, but flight testing was not performed.

Boeing B-737 horizontal stabilizer. Boeing's entry in medium primary structure development was the horizontal stabilizer on the B-737, the smallest airplane of their transport family. The stabilizer, shown in figure 8, is the two-spar torque box structure widely used in transport aircraft. The stabilizer is connected to carry-through structure by pin joints at the side of the fuselage. This design feature was particularly fortuitous because it allowed the composite stabilizer to be designed as a straightforward replacement item for the standard aluminum alloy stabilizer. Boeing pursued a conservative approach by fabricating cover panels, spars, and ribs as subassemblies and joining them with mechanical fasteners. The design, development, testing, and fabrication are summarized in reference 13.

The composite stabilizer experienced a structural failure during fail-safe tests required for FAA
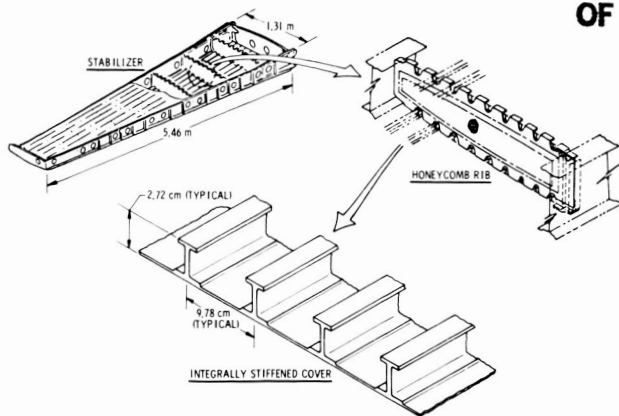
Figure 8. Graphite-epoxy horizontal stabilizer on B-737.

Figure 10. Graphite-epoxy vertical stabilizer on DC-10.

FAA CERTIFICATION AUGUST 1982 | REVENUE SERVICE INAUGURATION APRIL 11, 1984

Figure 9. Advanced composite horizontal stabilizer installed on B-737 for airline service.
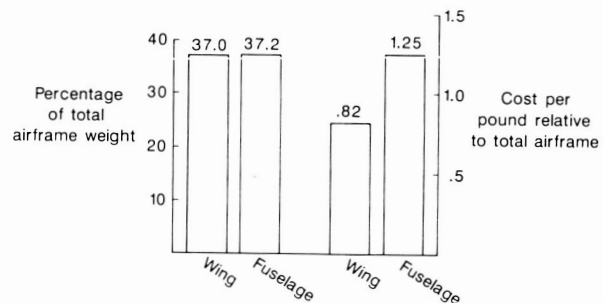
Figure 11. Relative weight and cost of metal transport wing and fuselage components.

certification (see ref. 12). After deficiencies were corrected, the stabilizer was certificated for flight service. Full-scale test results and production data from the five shipsets manufactured for commercial service are given in reference 14.

Figure 9 depicts a major milestone in the ACEE Program, the date when the composite stabilizers entered airline service. After years of effort, composite primary structures were a practical reality!

Douglas DC-10 vertical stabilizer. The vertical stabilizer on the DC-10 was the only Douglas structure that met technical and cost considerations for ACEE development, but the component afforded considerably less than optimum opportunities to demonstrate the advantages of composite materials. The metal stabilizer is a four-spar design with minimum gage skin panels. As a retrofit structure, the composite stabilizer was constrained to existing substructure and attachment interfaces. In addition, Douglas opted for a nonbuckling composite struc-

ture. With these constraints, achieving significant weight savings was a formidable challenge that led to the complex design shown in figure 10 and described in reference 15. The stabilizer spars and ribs were joined in a complicated secondary bonding operation to avoid the weight of metal fasteners. The honeycomb skin panels were bolted to the spar-rib substructure.

Development of the DC-10 composite stabilizer took considerably more time than expected. Fabrication problems and a test failure (see ref. 12) occurred. Nevertheless, the major goals have been achieved, and the development was successfully accomplished. The stabilizer received FAA certification in 1986 and entered commercial flight service in January 1987.

Wing and Fuselage Technology

From the outset, the goal of the ACEE Program was to develop and validate composite wing and fuselage structures. Figure 11 shows that these structures comprise about three-fourths of the aircraft structural weight, and thus weight savings in these components could significantly reduce fuel

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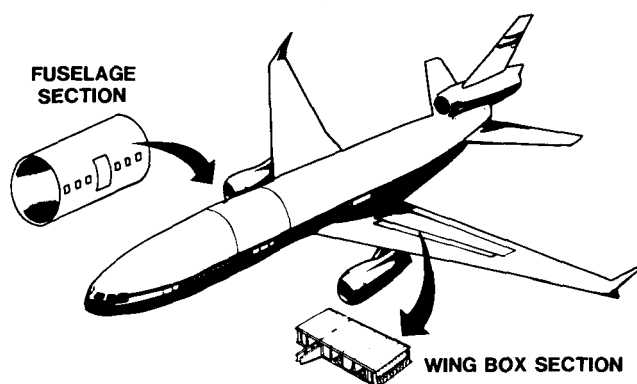
Figure 12. Large composite primary aircraft structures.

usage. The high relative cost of the metal fuselage dictated a strategy of using composites to reduce manufacturing cost. For the wing, the relative cost of metal would be hard to better, but the potential weight savings from composites could offset a higher relative cost and, thus, make composites economically viable.

When the ACEE Program began, development of a full-scale composite wing was expected to be underway by 1980, and development of a composite fuselage shortly thereafter. Events did not unfold as planned. The unanticipated issue of carbon fiber risk to electrical systems (ref. 16) required time and money to resolve. (See section entitled "Carbon Fiber Risk Assessment.") Also, experience with the medium primary components indicated that improvements were needed in design and analysis of composite structures and in the composite materials themselves. The latter issue triggered an intensive development of new toughened composite materials by industry and government. Under ACEE auspices, standard tests (ref. 17) and a specification (ref. 18) were established for toughened composite materials. The Langley Research Center instituted focused research (ref. 19) on tough composite materials.

Despite setbacks, NASA development of composites technology for transport wing and fuselage structures began in 1981. However, the approach differed from that followed for secondary and medium primary structures. Instead of designing and manufacturing full-size components as direct replacements for metal, the program focused on smaller but full-scale segments as shown in figure 12. Thus, technology validation would be achieved by short-span wing boxes and fuselage barrel sections.

For a composite wing, preparations began in 1978 with design trade-off studies by Boeing, Douglas, and Lockheed (see bibliography for reports). The first step in the actual development was to address long-lead-time key technology issues. Figure 13 depicts the investigations performed, which achieved considerable success. Douglas devised joint designs for heavily loaded wing structures and developed appropriate analysis methods. In demonstration tests, the joints achieved a strain to failure of 0.005, a significant improvement over existing designs. Lockheed addressed the system requirements for "wet wing" such as fuel containment and lightning protection. Methods to prevent fuel leakage and lightning effects degradation were demonstrated. Boeing devised wing panels that achieved post-impact compression stresses of 50 000 psi at strains of 0.006. These panels incorporated innovative damage limiting features in the skin and stiffeners. Also, a repair investigation was performed. New toughened resin composite materials were used and evaluated by the contractors. Data on these materials are included in contractor reports listed in the bibliography.

Having achieved success in the wing key technology contracts, NASA proceeded with the second phase: the development and demonstration of large-scale components. Major programs to design, fabricate, and test wing box components were begun with Lockheed and Douglas in 1984. One such component is shown in figure 14. References 20 and 21 describe the two programs. However, in early 1985, NASA deleted future contract funds and thereby terminated composite wing development under the ACEE Program.

Concurrent with the wing technology work, NASA began to develop technology for composite fuselage applications. Studies completed in 1984 identified major technology voids and areas of concern. Following these studies, contract work began to address the specific design issues of damage tolerance (Boeing), impact dynamics and acoustic transmission (Lockheed), and large cutouts (Douglas). The Boeing contract included a second phase which was to involve design, fabrication, and testing of full-scale fuselage panels. However, with ACEE Program funds deleted, fuselage development was discontinued. Work completed by Boeing, Lockheed, and Douglas is reported in references 22, 23, and 24.

Langley Base Technology Program

For decades the Langley Research Center has engaged in materials and structures research. Analysis methods and designs developed at Langley for metal structures are widely used in the aerospace industry. Likewise, Langley has played a leading role in the development of composite materials and their application to aerospace structures. In contrast to the major component developments pursued under the ACEE Program, the Langley program is focused on basic or generic technology development. The multidisciplinary nature of this base program is depicted in

BEFORE TEST | AFTER TEST

BOLT - 4
BOLT - 3
BOLT - 2
BOLT - 1

FRONT VIEW   SIDE VIEW

(a) Critical joints.

LIGHTNING STRIKE TEST

IMPACT TEST   FUEL LEAK TEST

RESIDUAL
STRENGTH
TEST

FATIGUE TEST

(b) Fuel containment and damage tolerance.

(c) Durability and damage tolerance.

Figure 13. Key technology issues in development of composite wing structures.

GOALS
35% weight loss
10% cost savings

Skin splice at sweep break
Gear load distribution
Pylon load distribution rib
Spar cap/web splice and trap panel
Landing gear load introduction
Aero break in lower surface

Aircraft features
- New design
- 150 passengers
- Supercritical airfoil
- 21° sweep
- 12.1 aspect ratio

Wing design features
- High wing loading (30,000 lb/in)
- Eccentric load path
  - Sweep break
  - Aero break
  - Dihedral box
- Side of body interface

Figure 14. Advanced composite inboard wing component for ground testing.

The results of this research are presented in a large number of reports covering a range of subjects.

These reports are presented in the annotated bibliography under "Langley Base Technology." The reports are grouped into subsections such as design and analysis or fatigue and fracture. It is impractical to present results in this paper from such a large and diverse data source. Therefore, the following paragraphs present brief summaries of the scope and direction of Langley research. The intent is to provide a reference as to what was done, who did it, and where results are published.

Structural Design, Analysis, and Evaluation

Design, test, and analysis of elements, particularly panels, are fundamental activities in structures research. Composite panels have, therefore, been the focus for Langley research, and the test data (fig. 16) have been compared with metal panel data acquired over many years. The figure of merit in such comparisons is weight saved. Reference 25 is typical of the many reports available on composite element evaluation.

Design synthesis (ref. 26) and analysis (ref. 27) involving computer codes are important features of the Langley research. Structural optimization codes developed at Langley, such as PASCO, are available for composite design. Computer codes have been applied to the design of curved stiffened panels (fig. 17) which have application to fuselage structure. In these designs, the skin between adjacent stiffeners is permitted to buckle under normal operating loads.

Damage Tolerance

In contrast to metals, carbon fiber composites are essentially brittle materials; that is, they do not yield

figure 15. Within the various disciplines, the research is performed by Langley scientists and by specialists working under grants and contracts.

Intense research in composites was performed by Langley during the ACEE era from 1976 to 1985. Impetus for this research came from the ACEE Program and from major development efforts by the Department of Defense. Cooperative research on helicopter structures was conducted with the U.S. Army. During those years, various universities obtained support and guidance from Langley in wide-ranging research.
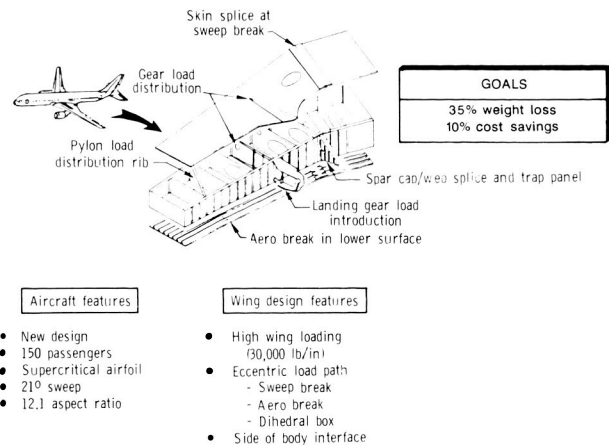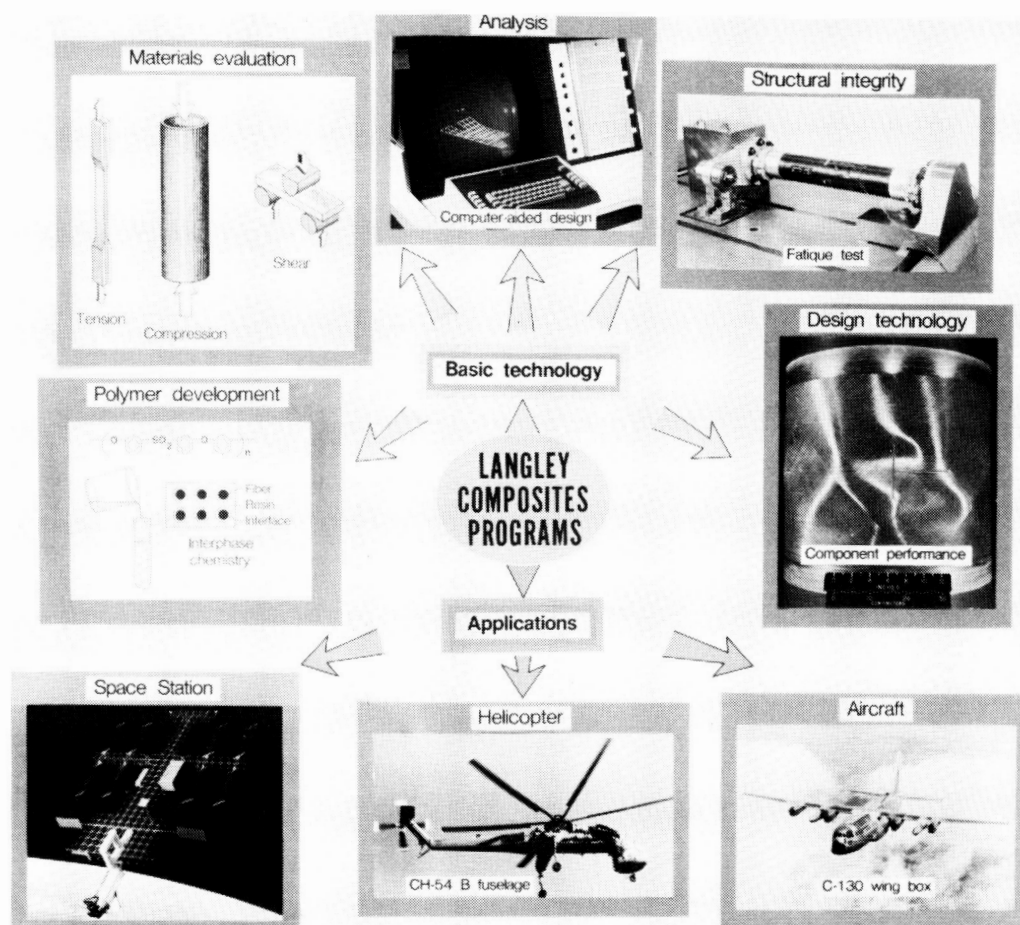
Figure 15. Multidisciplinary basic composites research programs at Langley.
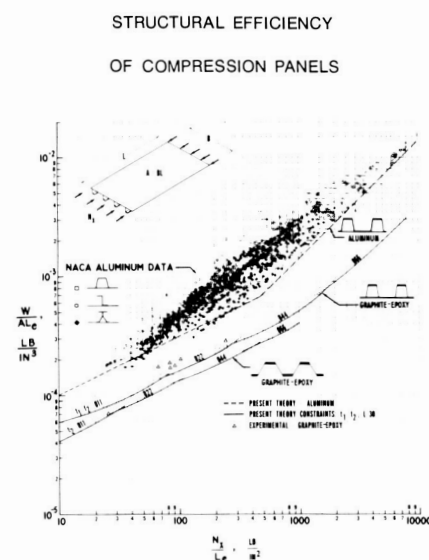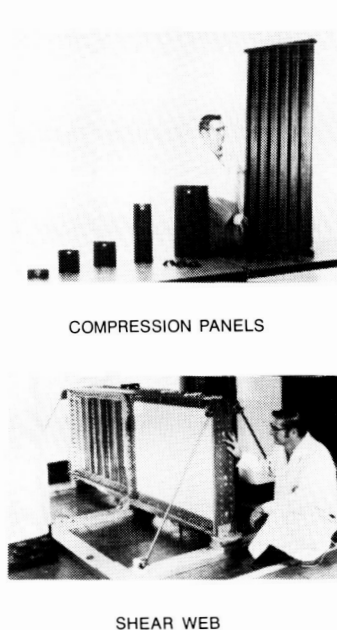

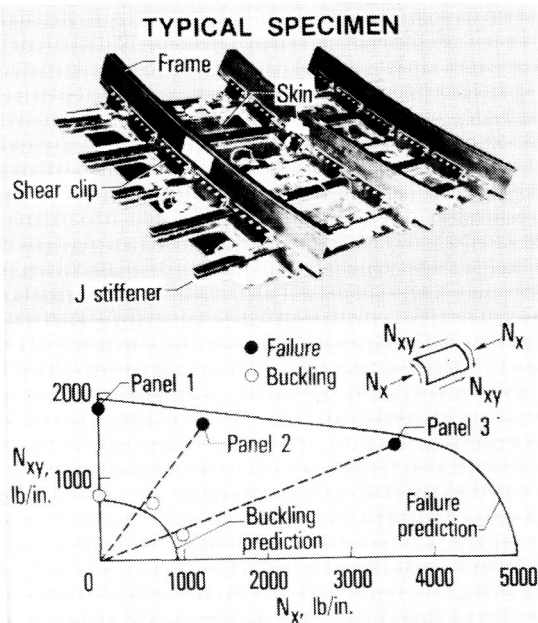Figure 16. Comparison of composite panels with metal panels.

Figure 17. Postbuckling capability of stiffened graphite-epoxy curved panels subjected to combined loads.

at local regions of high stress concentration which occurs around notches and cutouts or at impact damage sites. With the application of composites to heavily loaded primary structures, damage tolerance became a critical issue. Structures which have suffered either invisible or barely visible damage must be designed to be adequately strong. Methods to contain damage (ref. 28) and to minimize its effects (ref. 29) have been investigated at Langley and elsewhere. Also, the damage tolerance issue spurred efforts to develop new composite materials with improved toughness (see fig. 18).

Fatigue, Fracture, and Delamination Mechanics

Fatigue and fracture of structures are other fundamental research activities that have been conducted at Langley for many years. As composite materials became increasingly important, the focus of the research shifted from metals to composites.

The classical issues of fatigue and fracture (ref. 30) have been pursued under the Langley program. As depicted in figure 19, the research is aimed at obtaining a fundamental understanding of material behavior under cylic loading. The results are expected to guide structural applications and new material developments.

In addition to the classical failure mechanisms, composites are subject to failure by delamination. Delamination analyses (ref. 31) and evaluation (ref. 32) have received considerable emphasis in the Langley base technology program.

Environmental Effects and Flight Service

Environmental durability was recognized early in the Langley program as a key issue in the future application of composite materials. The aircraft industry had experienced material durability problems (such as corrosion and stress corrosion) and would, therefore, be conservative in adopting yet another new material. To provide data to resolve the issue, extensive investigations were begun to determine how composite materials reacted to various fluids, sunlight, and ground and flight environments.

The investigations have involved specimens ranging from small coupons (fig. 20) to full-scale aircraft and helicopter components (fig. 21). Specimen and component exposures have been conducted on a world-wide basis. Analytical methods have been developed to predict material response to moisture with and without sunlight. Environmental effects data (ref. 33) and compilations of flight service experience (ref. 34) are reported periodically.

The flight service data now total millions of flight hours. (See table 1 for the latest compilation.) These data have been instrumental in establishing confidence in the application of composites to aircraft and helicopter structures.

Material Processing, Properties, and Test Methods

The material processing research performed in the Langley base program includes processing science and fundamental studies of resins and fibers. The processing science involves process chemistry, rheology, and cure mechanics (ref. 35). Novel material forms and fabrication methods (ref. 36) have been developed in laboratory-scale studies and experiments. Full-scale manufacturing tools and methods are not developed in the Langley program; information on such activities is available in ACEE documents or from Department of Defense (DOD) contracts.

The compilation of material properties or the development of design data is not a major element of the Langley base program. The Langley approach has been to develop technology rather than extensive data bases. Nonetheless, about 19 documents containing material property data are cited in the bibliography. Also, NASA has provided funds for the *DOD/NASA Advanced Composites Design Guide* (ref. 37). ACEE component data are included in the *DOD/NASA Structural Composites Fabrication Guide* (ref. 38).

Test methods have been developed as necessary adjuncts to experimental research in materials and structures. Over 40 documents dealing with test methods are cited in the bibliography. Tests

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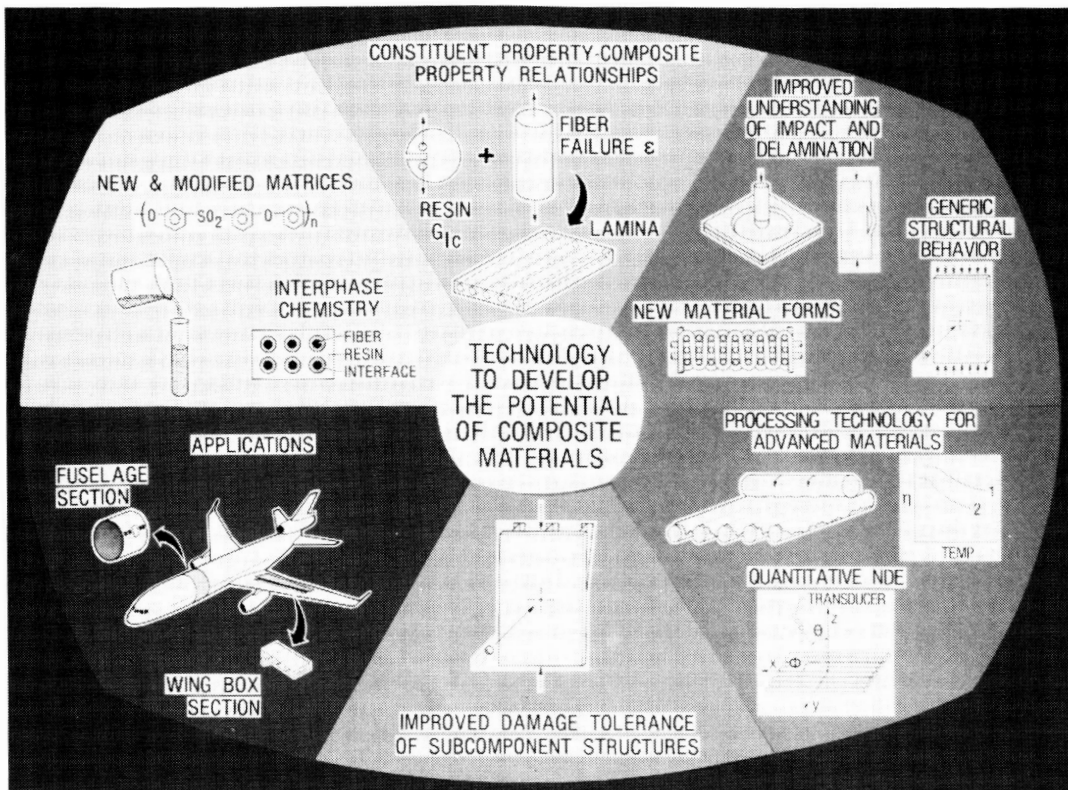
Figure 18. Program to improve toughness of composite materials.
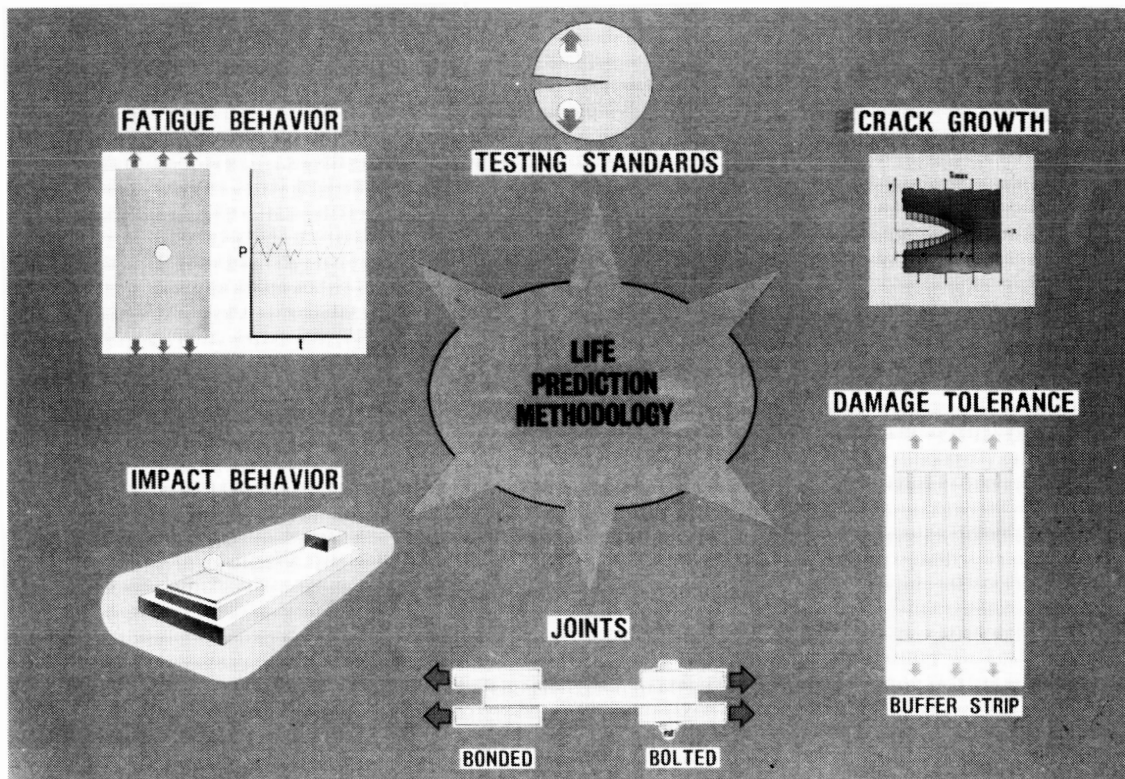
Figure 19. Fatigue analysis of composites.
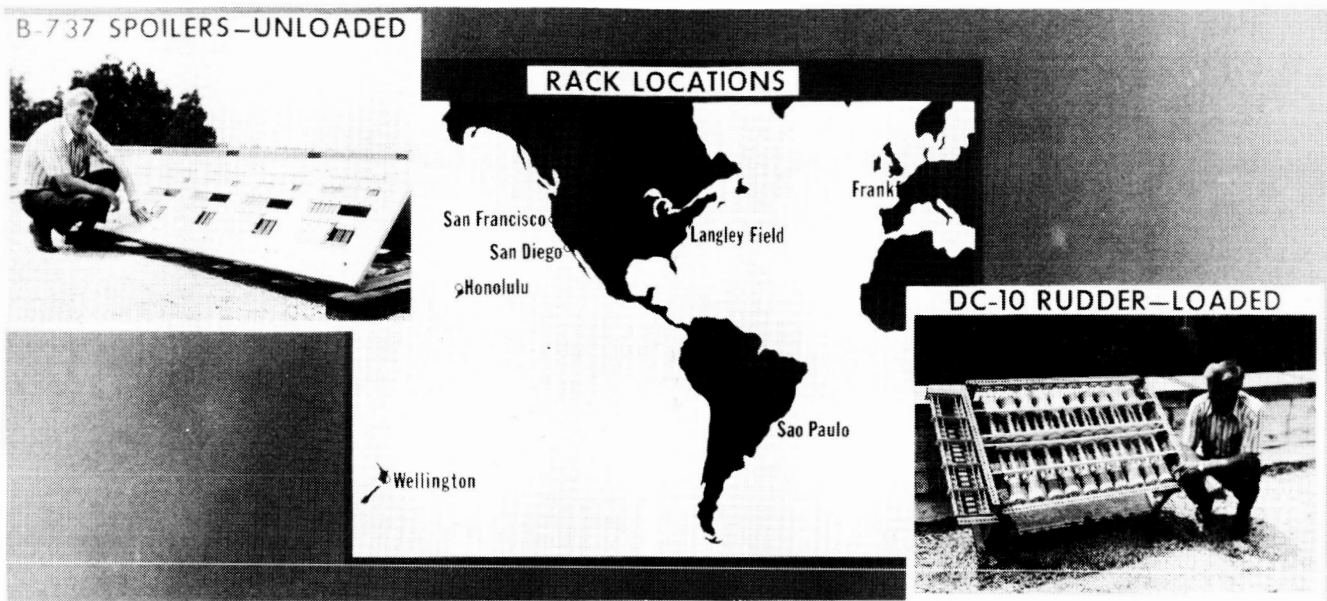
Figure 20. Environmental exposure of composite components used in flight service evaluations.
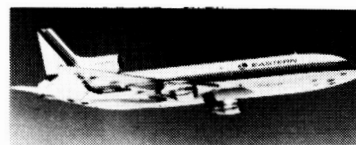
L-1011 FAIRING

B-737 SPOILER

CH-54 B FUSELAGE

206 L DOORS AND FAIRING

DC-10 RUDDER AND AFT PYLON

C-130 WING BOX

Figure 21. Flight service evaluation of composite structural components.

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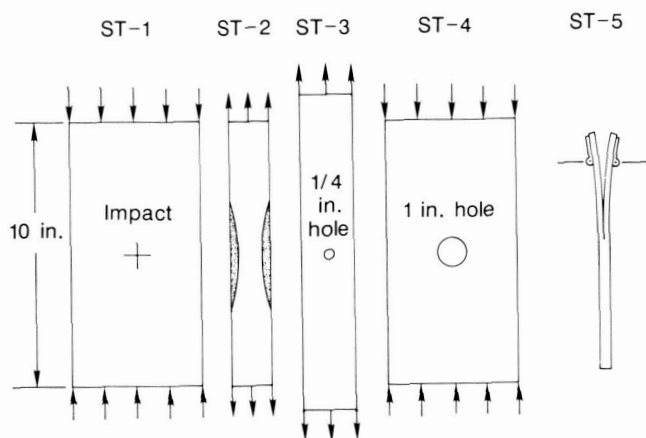
Figure 22. Standard tests for toughened resin composites.
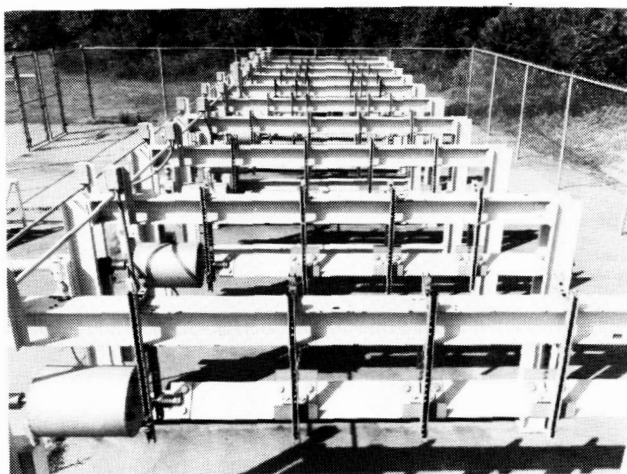
Figure 23. Test machines for outdoor test of wing joints.

Figure 24. Acoustic testing of a composite cylinder.

developed by NASA for evaluating composite materials are shown in figure 22 and described in reference 17.

Joints and Cutouts

Research on joints and cutouts has ranged from sophisticated finite element analyses (ref. 39) to simple strength tests (ref. 40). Data have been obtained for both bolted and bonded joints and for various materials: graphite-epoxy, graphite-polyimide, and Du Pont Kevlar-epoxy. Specialized test machines (see fig. 23) have been used to obtain long-term durability data on wing-skin splices subjected to outdoor exposure under load.

Impact Dynamics and Acoustics

Experimental and analytical efforts are underway at Langley to investigate the response of composite structures under crash loading conditions. Among the areas being addressed are energy absorption,

tearing of skin panels, friction and abrasion of composite materials, and dynamic response of structural elements. A discussion of the research underway is presented in reference 41, while the results of some fundamental research in energy absorption are given in reference 42.

Acoustic effects are anticipated to be a major issue in the application of composites to fuselage structures. Research (ref. 43) is underway to characterize the materials and to determine the response of typical structures. Figure 24 shows a filament-wound composite cylinder produced under the ACEE Program (ref. 23) undergoing acoustic testing at Langley.

Graphite-Polyimide and Polymer Development

In addition to research involving epoxy-matrix composites, Langley has worked for years on high-temperature polymer-matrix materials. These materials include polyimides, thermoplastics, and other novel polymers that are processed at higher temperatures than the 350°F maximum typical of structural epoxies. Initial impetus to this research was provided by a project called Composites for Advanced Space Transportation Systems (CASTS) described in reference 44. This work, which began in 1975, was to develop graphite-polyimide structures with 600°F operational capability for application to vehicles such as the Space Shuttle.

More recently the high-temperature polymer research has focused on advanced polymer synthesis (fig. 25) and characterization methodology to develop novel polymeric materials. The new materials are for applications such as matrices for composites, adhesives, and films for spacecraft (refs. 45 and 46).

Materials and Structures for Helicopters

The helicopter composites program at Langley is a joint effort by NASA and the U.S. Army Aerostructures Directorate. The research is focused on basic technology rather than on major hardware development, which the Army pursues elsewhere. The

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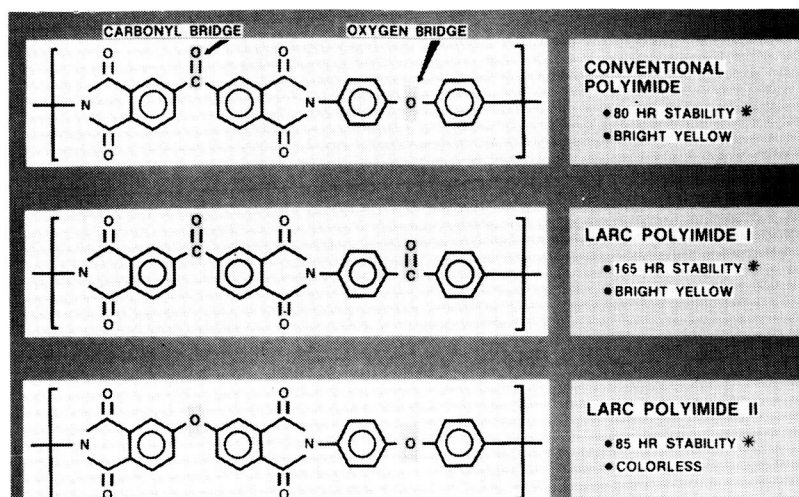
Figure 25. Tailoring of polymer structures to control polymer properties.

FORWARD FAIRING

Kevlar-epoxy fabric
stiffened foam sandwich
mass (kg) 2.31
size (m) 0.90 x 0.74

VERTICAL FIN

Graphite-epoxy tape
fiber truss core
mass (kg) 5.90
size (m) 1.98 x 0.50

LITTER DOOR

Kevlar-epoxy fabric
two skins—hollow section
mass (kg) 3.72
size (m) 1.17 x 0.66

BAGGAGE DOOR

Kevlar-epoxy fabric
honeycomb sandwich
mass (kg) 1.41
size (m) 0.97 x 0.58
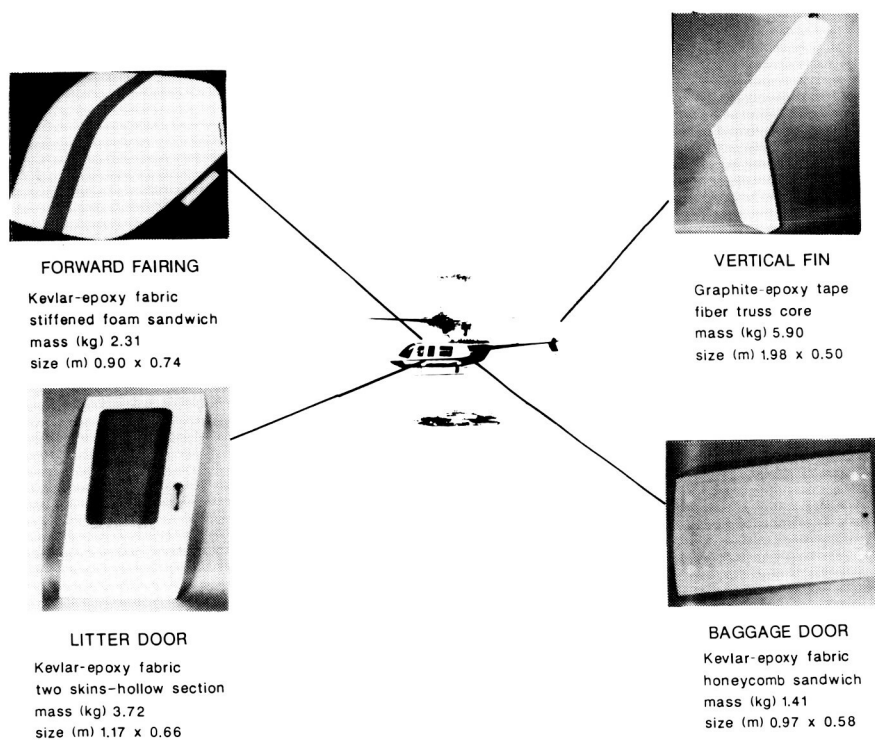
Figure 26. Composite components for flight service evaluation on 206L helicopter.
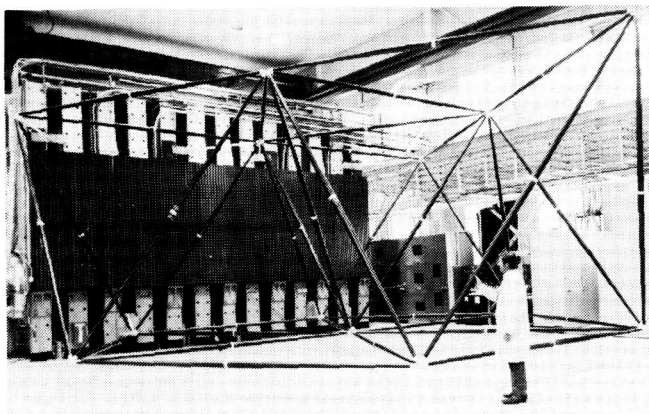
Figure 27. Composite truss for space applications.

NEEDS
- SOLID DATA BASE
- VALIDATED DESIGN/ANALYSIS PROCEDURES
- INNOVATIVE MANUFACTURING
- LOW COST MATERIALS
- ADVANCED MATERIAL FORMS

BENEFITS
- 25-35% STRUCTURAL WEIGHT REDUCTION
- 10-25% ACQUISITION COST SAVINGS
- 12-15% FUEL SAVINGS
- GOOD DURABILITY
- LOW MAINTENANCE

Figure 28. Benefits of "all composite" transport aircraft.

Langley program has developed components for flight service evaluation (fig. 26) and selected structural elements.

General Aviation Research

Although the technology development in the ACEE and Langley base programs is applicable to general aviation aircraft, there are few instances of composites research specifically directed to general aviation aircraft. Plans were generated periodically for an increased focus on research in structures and materials for general aviation, but these plans were never implemented.

Repair Methods

The Langley base program has not emphasized all areas of composites research. Repair methods have been limited to generic investigations cited in the bibliography. However, research on bonding and adhesives cited under "Materials Development and Processing" (see ref. 36) has direct application to repair of composites. A recent investigation of repaired laminates after outdoor exposure under load is reported in reference 47. These laminates were exposed in the specialized machines shown in figure 23.

Space Applications

Composite materials have desirable attributes for use in space. Low mass is obviously important. In addition, composites possess thermal and dimensional stability. Langley has extensively investigated Space Station truss members fabricated from graphite fiber composites. Figure 27 shows a composite truss assembled for ground testing. Considerable research has been conducted on the effect of space radiation on resin-matrix composites (see ref. 48).

Carbon Fiber Risk Assessment

The potential risk to electrical systems posed by the accidental release of carbon fibers became an issue in the late 1970's. NASA, one of several agencies involved in a risk assessment, was responsible for the aviation aspects of the potential problem. The Langley Research Center directed an intense and comprehensive analytical and experimental investigation which showed conclusively (ref. 16) that the risk was insignificant.

Concluding Remarks

The period from 1975 to 1986 was the golden age of composites research in the United States. Exciting, almost revolutionary, developments were achieved in composite materials and structures. Military aircraft now routinely employ composite structures and expanded applications on new aircraft are certain. In commercial transports, applications are not as aggressive, but secondary structures are in production for airplanes such as the Boeing B-757 and B-767 and the Douglas MD-80.

During this period, the development and exploitation of composites technology became a national effort involving scientists and engineers in government, industry, and universities. The ACEE Program served as a strategic center of gravity providing the national effort with coherence and clear objectives. Other major elements in the total effort, such as Langley's base research, drew upon the ACEE Program for funding or focus.

The ACEE Program was terminated prior to accomplishment of its major goals. No commitment has yet been made to use composite primary structures in transport aircraft, either military or commercial. The benefits listed in figure 28 that accrue for a

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composite transport appear feasible, but the critical issues of affordable technology remain unanswered. The Langley base program continues to provide significant technology, but a base program alone will never resolve all the issues.

NASA Langley Research Center
Hampton, VA 23665-5225
May 22, 1987

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38. Meade, L. E. "Roy": *DOD/NASA Structural Composites Fabrication Guide. Volumes I and II, 3rd ed.:* AFWAL-TR-85-4107, Vols. I & II, U.S. Air Force, Sept. 1982. (Available from DTIC as AD B096 021L & AD B096 022L.)

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Table 1. NASA Composite Structures Flight Service Summary

[Summary as of March 1987]

Aircraft component	Total no. of components (a)		Start of flight service	Cumulative flight hours	
				Highest time on one aircraft	Total
L-1011 fairing panels	18	(15)	January 1973	39 210	599 240
B-737 spoiler	108	(47)	July 1973	38 560	2 380 550
C-130 center wing box	2	(2)	October 1974	8 340	16 430
DC-10 aft pylon skin	3	(2)	August 1975	33 240	83 440
DC-10 upper aft rudder	[b]15	(12)	April 1976	39 400	350 210
B-727 elevator	10	(8)	March 1980	21 980	184 940
L-1011 aileron	8	(8)	March 1982	16 890	128 060
B-737 horizontal stabilizer	10	(10)	March 1984	8 850	79 040
DC-10 vertical stabilizer	1	(1)	January 1987	—	—
S-76 tail rotors and horizontal stabilizer	14	(3)	February 1979	5 800	52 100
206L fairing, doors, and vertical fin	160	(84)	March 1981	7 300	330 000
CH-53 cargo ramp skin	1	(1)	May 1981	1 600	1 600
Total	350	(193)			4 205 610

[a]Numbers in parentheses indicate number of components still in service.
[b]Five more rudders to be installed.

Appendix

Bibliography of Composites Reports

The technical publications listed herein (with their abstracts) were generated in conjunction with the ACEE Program and the Langley Research Center base composites technology program. Within these two main categories, the documents are grouped into subsections according to subject matter and listed alphabetically by author. The subsection titles, given in the "Contents," parallel the subject areas discussed in the text. An index by author is provided to increase the usefulness of this compilation.

The publications listed deal only with resin-matrix fibrous composite materials and with aerospace applications. Metal-matrix and boron-epoxy materials are excluded.

The abstracts used are from the NASA Scientific and Technical Information System. License was taken to modify or shorten abstracts. Accession numbers, report numbers, and other identifying information are included in the citations to facilitate filling of requests for specific documents.

Availability sources of the different types of materials are as follows:

Availability

Accession number	Type of material	Where obtained
00A00000 Example: 75A-25583	AIAA paper and published literature available from AIAA or in journals, conferences, etc., as indicated	American Institute of Aeronautics and Astronautics Technical Information Service 55 West 57th Street, 12th Floor New York, NY 10019
00N00000 Example: 67N-37604	Report literature having no distribution limitation	National Technical Information Service (NTIS) 5285 Port Royal Road Springfield, VA 22161
00X00000 Example: 72X76040	Report literature having some type of distribution limitation	NASA Scientific and Technical Information Facility (STIF) P. O. Box 8757 B.W.I. Airport, MD 21240

A "#" after an accession number indicates that the document is also available in microfiche form.

ACEE Composites Program

ACEE overviews and summaries

1. Bohon, H. L.; Chapman, A. J., III; and Leybold, H. A.: *Ground Test Experience With Large Composite Structures for Commercial Transports.* NASA TM-84627, Mar. 1983. (Presented at the American Helicopter Society Composite Structures Specialist's Conference, Philadelphia, Pa., Mar. 23–25, 1983.)

83N29732#

The initial ground test of each component resulted in structural failure at less than ultimate design loads. While such failures represent major program delays, the investigation and analysis of each failure revealed significant lessons for effective utilization of composites in primary structure. Foremost among these are secondary loads that produce through-the-thickness forces which may lead to serious weaknesses in an otherwise sound structural design.

2. Bohon, H. L.; and Davis, J. G., Jr.: Composites for Large Transports—Facing the Challenge. *Aerospace America*, vol. 22, June 1984, pp. 58–62.

84A36797#

NASA has undertaken development and test programs in order to remove existing barriers to the use of composite material primary structures and to assess their advantages in terms of both acquisition cost and mission performance. These programs are expected to reach design technology readiness for wing and fuselage structures by 1988.

3. James, R. L., Jr.; and Maddalon, D. V.: *Airframe Technology for Aircraft Energy Efficiency.* NASA TM-85749, Mar. 1984.

84N18154#

The economic factors that resulted in the implementation of the Aircraft Energy Efficiency Program (ACEE) are reviewed and airframe technology elements including content, progress, applications, and future direction are discussed. The program includes the development of laminar flow systems, advanced aerodynamics, active controls, and composite structures.

4. James, R. L., Jr.; and Maddalon, D. V.: The Drive for Aircraft Energy Efficiency. *Aerospace America*, vol. 22, Feb. 1984, pp. 54–58.

84A29569#

NASA's Aircraft Energy Efficiency (ACEE) Program, which began in 1976, has mounted a development effort in four major transport aircraft technology fields: laminar flow systems, advanced aerodynamics, flight controls, and composite structures. Composite structures yield lighter airframes that in turn call for smaller wing and empennage areas, reducing induced drag for a given payload. In combination, all four areas of development are expected to yield a fuel consumption reduction of 40 percent.

5. Leonard, R. W.: *Fuel Efficiency Through New Airframe Technology.* NASA TM-84548, Aug. 1982.

85N10909#

In its Aircraft Energy Efficiency Program, NASA has expended approximately 200 million dollars toward development and application of advanced airframe technologies to United States commercial transports. United States manufacturers have already been given a significant boost toward early application of advanced composite materials to control surface and empennage structures and toward selected applications of active controls and advanced aerodynamic concepts. In addition, significant progress in definition and development of innovative but realistic systems for transports has already been made.

6. Leonard, R. W.; and Mulville, D. R.: Current and Projected Use of Carbon Composites in United States Aircraft. Paper presented at AGARD Specialists Meeting on Electromagnetic Effects of Carbon Composite Materials Upon Avionics Systems, Lisbon, Portugal, June 16–19, 1980.

80A34840#

Carbon composite materials are beginning to be used in commercial transports, general aviation aircraft, military fighter aircraft, and helicopters due to demonstrated weight savings and potential manufacturing cost savings. Attention is given to current production applications of carbon composites which range from the secondary structures of new commercial transports to wing primary structures of fighters. Current development efforts are discussed that will lead to their future application to fuselages, as well as whole airframes. Finally, laminate constructions, which vary widely, and may be relevant to avionics system design, are examined.

7. Leonard, R. W.; and Wagner, R. D.: Airframe Technology for Energy Efficient Transport Aircraft. SAE Paper 760929, 1976.

77A28234

NASA has initiated a comprehensive Aircraft Energy Efficiency Program which is concerned with the development of approaches for reducing fuel consumption in new aircraft of the 1980–2000 time period. A review is presented of the airframe technologies selected for emphasis in the NASA program, taking into account an evaluation of their potential for reducing transport direct operating costs through fuel efficiency. Attention is given to the importance of fuel efficiency, the impact of advanced technology, advanced composite structures, the NASA composite primary structures program, advanced aerodynamics and active controls, supercritical wing geometry, active load and flutter control, and aspects of laminar flow control.

8. Leonard, R. W.: Airframes and Aerodynamics. *Astronautics and Aeronautics,* vol. 16, July–Aug. 1978, pp. 38–46.

78A43359#

The first part of the paper discusses the Energy Efficient Transport Program of the Aircraft Energy Efficiency (ACEE) Program, giving attention to the development of active aerodynamics and active controls. The second part of the paper deals with two other portions of the ACEE Program: Composite Primary Structures and Laminar Flow Control.

9. Maddalon, D. V.; and Wagner, R. D.: Energy and Economic Trade Offs for Advanced Technology Subsonic Aircraft. *Proceedings of the 4th Annual Intersociety Conference on Transportation* (Los Angeles, Calif., July 18–23), ASME, 1976.

77A29471

Changes in future aircraft technology which conserve energy are studied, along with the effect of these changes on economic performance. Among the new technologies considered are laminar-flow control, composite materials with and without laminar-flow control, and advanced airfoils. Aircraft design features studied include high-aspect-ratio wings, thickness ratio, and range. Engine technology is held constant at the JT9D level. It is concluded that wing aspect ratios of future aircraft are likely to significantly increase as a result of new technology and the push of higher fuel prices. Whereas current airplanes have been designed for AR = 7, supercritical technology and much higher fuel prices will drive aspect ratio to the AR = 9–10 range. Composite materials may raise aspect ratio to about 11–12 and practical laminar–flow-control systems may further increase aspect ratio to 14 or more. Advanced technology provides significant reductions in aircraft take-off gross weight, energy consumption, and direct operating cost.

10. Nagel, A. L.; Alford, W. J., Jr.; and Dugan, J. F., Jr.: *Future Long-Range Transports: Prospects for Improved Fuel Efficiency.* NASA TM X-72659, Feb. 1975.

75N17339#

A status report is provided on current thinking concerning potential improvements in fuel efficiency and possible alternate fuels. Topics reviewed are (1) historical trends in airplane efficiency; (2) technological opportunities including supercritical aerodynamics, (3) vortex diffusers, (4) composite materials, (5) propulsion systems, (6) active controls, and terminal-area operations; (7) unconventional design concepts; and (8) hydrogen-fueled airplanes.

11. Vosteen, L. F.: Composite Structures for Commercial Transport Aircraft. Paper presented at the American Institute of Mining, Metallurgical and Petroleum Engineers, 2nd International Conference on Composite Materials, Toronto, Canada, Apr. 16–20, 1978.

78A34901#

A description is provided of a NASA-industry program which will lead to a more extensive use of advanced composites. The approach being taken is to develop, for existing aircraft, components which have potential for significant weight savings. The design of each of the considered components is described, taking into account a graphite-epoxy rudder, an inboard aileron, a graphite-epoxy elevator, a vertical tail, a horizontal tail, and a vertical stabilizer.

12. Vosteen, L. F.: *Composite Structures for Commercial Transport Aircraft.* NASA TM-78730, June 1978.

78N27183#

The development of graphite-epoxy composite structures for use on commercial transport aircraft is considered. Six components, three secondary structures, and three primary structures, are presently under development. The six components are described along with some of the key features of the composite designs and their projected weight savings.

13. Vosteen, L. F.: Composite Aircraft Structures. *Fibrous Composites in Structural Design,* Plenum Press, 1980, pp. 7–24.

82A27127

In late 1975, the Aircraft Energy Efficiency (ACEE) Program was initiated by NASA in order to accelerate the development of selected technologies which showed promise for substantial improvements in the fuel efficiency of commercial transport aircraft. A description is presented of the status of the composite structure development programs which form one of the six sections of the ACEE Program. Six aircraft components are currently being developed under NASA contract by three major transport manufacturers. The components include the upper aft rudder of the Douglas DC-10, the inboard ailerons of the Lockheed L-1011, the elevators of the Boeing 727, the vertical stabilizers for the Lockheed L-1011 and Douglas DC-10, and the horizontal stabilizers of the Boeing 737.

14. Wright, H. T.: NASA Technology Program for Future Civil Air Transports. AIAA Paper 83-1603, 1983.

83A33366

An assessment is made of the development status of technology, applicable to future civil air transport design, which is currently undergoing conceptual study or testing at NASA facilities. The NASA civil air transport effort emphasizes advance aerodynamic computational capabilities, fuel-efficient engines, advanced turboprops, composite primary structure materials, advanced aerodynamic concepts in boundary layer laminarization and aircraft configuration, refined control, guidance and flight management systems, and the integration of all these design elements into optimal systems. Attention is given to such novel transport aircraft design concepts as forward swept wings, twin fuselages, sandwich composite structures, and swept blade propfans.

ACEE transport structures development

15. Alva, T.; Henkel, J.; Johnson, R.; Carll, B.; Jackson, A.; Mosesian, B.; Brozovic, R.; O'Brien, R.; and Eudaily, R.: *Advanced Manufacturing Development of a Composite Empennage Component for L-1011 Aircraft*. NASA CR-165885, May 1982.

85N11141#

This is the final report of technical work conducted during the fourth phase of a multiphase program. The empennage component selected for this program is the vertical fin box of the L-1011 aircraft. The box structure extends from the fuselage production joint to the tip rib and includes front and rear spars. During Phase 4 of the program, production quality tooling was designed and manufactured

to produce three sets of covers, ribs, spars, miscellaneous parts, and subassemblies to assemble three complete ACVF units. Recurring and nonrecurring cost data were compiled and documented.

16. Aniversario, R. B.; Harvey, S. T.; McCarty, J. E.; Parsons, J. T.; Peterson, D. C.; Pritchett, L. D.; Wilson, D. R.; and Wogulis, E. R.: *Full-Scale Testing, Production and Cost Analysis Data for the Advanced Composite Stabilizer for Boeing 737 Aircraft, Volume 2*. NASA CR-166012, Dec. 1982.

85N13915#

The development, testing, production activities, and associated costs that were required to produce 5 1/2 advanced-composite stabilizer shipsets for Boeing 737 aircraft are defined and discussed.

17. Aniversario, R. B.; Harvey, S. T.; McCarty, J. E.; Parsons, J. T.; Peterson, D. C.; Pritchett, L. D.; Wilson, D. R.; and Wogulis, E. R.: *Design, Ancillary Testing, Analysis and Fabrication Data for the Advanced Composite Stabilizer for Boeing 737 Aircraft, Volume 2*. NASA CR-166011, Dec. 1982.

85N13914#

Results of tests conducted to demonstrate that composite structures save weight, possess long term durability, and can be fabricated at costs competitive with conventional metal structures are presented with focus on the use of graphite-epoxy in the design of a stabilizer for the Boeing 737 aircraft. Component definition, materials evaluation, material design properties, and structural elements tests are discussed. Fabrication development, as well as structural repair and inspection, is also examined.

18. Aniversario, R. B.; Harvey, S. T.; McCarty, J. E.; Parsons, J. T.; Peterson, D. C.; Pritchett, L. D.; Wilson, D. R.; and Wogulis, E. R.: *Design, Ancillary Testing, Analysis and Fabrication Data for the Advanced Composite Stabilizer for Boeing 737 Aircraft. Volume 1: Technical Summary*. NASA CR-3648, Apr. 1983.

85N13913#

The horizontal stabilizer of the 737 transport was redesigned. Five shipsets were fabricated using composite materials. Weight reduction greater than the 20% goal was achieved. Parts and assemblies were readily produced on production-type tooling. Quality assurance methods were demonstrated. Repair methods were developed and demonstrated. Strength and stiffness analytical methods were substantiated by comparison with test results. Cost data were accumulated in a semiproduction environment. FAA certification was obtained.

19.   Aniversario, R. B.; Harvey, S. T.; McCarty, J. E.; Parsons, J. T.; Peterson, D. C.; Pritchett, L. D.; Wilson, D. R.; and Wogulis, E. R.: *Full-Scale Testing, Production and Cost Analysis Data for the Advanced Composite Stabilizer for Boeing 737 Aircraft. Volume 1: Technical Summary.* NASA CR-3649, Apr. 1983.

85N13912#

The full scale ground test, ground vibration test, and flight tests conducted to demonstrate a composite structure stabilizer for the Boeing 737 aircraft and obtain FAA certification are described. Detail tools, assembly tools, and overall production are discussed. Cost analyses aspects covered include production costs, composite material usage factors, and cost comparisons.

20.   Ary, A.; Axtell, C.; Fogg, L.; Jackson, A.; James, A. M.; Mosesian, B.; Vanderwier, J.; and Vanhamersveld, J.: *Flight Service Evaluation of an Advanced Composite Empennage Component on Commercial Transport Aircraft. Phase 1: Engineering Development.* NASA CR-144986, May 1976.

78N23080#

The empennage component selected for this program is the vertical fin box of the L-1011 aircraft. The box structure extends from the fuselage production joint to the tip rib and includes the front and rear spars. Various design options were evaluated to arrive at a configuration which would offer the highest potential for satisfying program objectives. The preferred configuration selected consists of a hat-stiffened cover with molded integrally stiffened spars, aluminum trussed composite ribs, and composite sandwich web ribs with integrally molded caps. Material screening tests were performed to select an advanced composite material system for the Advanced Composite Vertical Fin (ACVF) that would meet the program requirements from the standpoint of quality, reproducibility, and cost.

21.   Chovil, D. V.; Harvey, S. T.; McCarty, J. E.; Desper, O. E.; Jamison, E. S.; and Syder, H.: *Advanced Composite Elevator for Boeing 727 Aircraft. Volume 1: Technical Summary.* NASA CR-3290, Nov. 1981.

84N27722#

The design, development, analysis, and testing activities and results that were required to produce five and one-half shipsets of advanced composite elevators for Boeing 727 aircraft are summarized. During the preliminary design period, alternative concepts were developed. After selection of the best design, detail design and basic configuration improvements were evaluated. All program goals (except competitive cost demonstration) were accomplished when our design met or exceeded all requirements, criteria, and objectives.

22.   Chovil, D. V.; Grant, W. D.; Jamison, E. S.; Syder, H.; Desper, O. E.; Harvey, S. T.; and McCarty, J. E.: *Advanced Composite Elevator for Boeing 727 Aircraft, Volume 2.* NASA CR-159258, Nov. 1980.

83N16330#

Preliminary development efforts consisted of evaluating and selecting material, identifying ancillary structural development test requirements, and defining full scale ground and flight test requirements necessary to obtain Federal Aviation Administration (FAA) certification. After selection of the optimum elevator configuration, detail design was begun and included basic configuration design improvements resulting from manufacturing verification hardware, the ancillary test program, weight analysis, and structural analysis. Detail and assembly tools were designed and fabricated to support a full-scope production program, rather than a limited run.

23.   Cominsky, A.: *Manufacturing Development of DC-10 Advanced Rudder.* NASA CR-159060, Aug. 1979.

81N27077#

The design, manufacture, and ground test activities during development of production methods for an advanced composite rudder for the DC-10 transport aircraft are described. The advanced composite aft rudder is satisfactory for airline service and a cost saving in a full production manufacturing mode is anticipated.

24.   Dorward, F.; and Ketola, R. N.: Static and Damage Tolerance Tests of an Advanced Composite Vertical Fin for L-1011 Aircraft. *24th AIAA Structures, Structural Dynamics and Materials Conference* (Lake Tahoe, Nev., May 2–4), *Technical Papers, Part 1,* AIAA, 1983, pp. 516–527. (Available as AIAA Paper 83-0970.)

83A29780#

This paper recounts the significant events which took place during the structural verification testing of two graphite/epoxy material, full-size vertical stabilizers. The ground test articles were tested to a high bending dynamic lateral gust condition. The first

unit failed during static testing at 98 percent design ultimate load. Failure began within the front spar cap. A detailed review of the failure was performed to identify all possible modes. This review resulted in a "production line" type fix being designed for incorporation in the second ground test article prior to installation in the test fixture. The modified second unit sustained 106 percent of design ultimate load without incident.

25. Dunning, E. G.; Coobs, W. L.; and Legg, R. L.: *Advanced Composite Aileron Manufacture.* NASA CR-165718, June 1981.

83N30522#

The fabrication activities of the Advanced Composite Aileron program are discussed. These activities included detail fabrication, manufacturing development, assembly, repair, and quality assurance. Five shipsets of ailerons were manufactured.

26. Griffin, C. F.: Design Development of an Advanced Composite Aileron. AIAA Paper 79-1807, 1979.

79A47891#

This paper summarizes the design development of an advanced composite inboard aileron for the L-1011 commercial transport aircraft. Design details of the composite aileron are reported. Results of tests which substantiate the structural integrity of the design are also presented. The composite aileron is a multi-rib assembly with graphite/epoxy tape-syntactic core sandwich covers, a graphite/epoxy tape front spar, and graphite/epoxy fabric ribs. This structure is a direct replacement for the current metal aileron with a weight savings of 28.7 percent (40.3 lb.). Engineering cost estimates indicate that the composite structure will be cost competitive with the metal structure it is replacing.

27. Griffin, C. F.; Fogg, L. D.; and Dunning, E. G.: *Advanced Composite Aileron for L-1011 Transport Aircraft: Design and Analysis.* NASA CR-165635, Apr. 1981.

83N30406#

The aileron design is a multi-rib configuration with single piece upper and lower covers mechanically fastened to the substructure. Covers, front spar, and ribs are fabricated with graphite/epoxy tape or fabric composite material. The design has a weight savings of 23 percent compared to the aluminum aileron. The composite aileron has 50 percent fewer fasteners and parts than the metal aileron and is predicted to be cost competitive. Structural integrity of the composite aileron was verified by structural analysis and an extensive test program.

28. Griffin, C. F.: *Advanced Composite Aileron for L-1011 Transport Aircraft: Ground Tests and Flight Evaluation.* NASA CR-165664, Feb. 1981.

83N30405

A composite aileron and a metal aileron were subjected to a series of comparative stiffness and vibration tests. These tests showed that the stiffness and vibration characteristics of the composite aileron are similar to the metal aileron. The first composite ground test article was statically tested to failure which occurred at 139 percent of design ultimate load. The second composite ground test article was tested to verify damage tolerance and fail-safe characteristics.

29. Griffin, C. F.; and Dunning, E. G.: *Development of an Advanced Composite Aileron for the L-1011 Transport Aircraft.* NASA CR-3517, Feb. 1982.

84N27836#

Significant improvements in structural efficiency can be achieved by the utilization of advanced composites for construction of aircraft secondary structures. Careful evaluation of alternate designs and materials for the L-1011 advanced composite inboard aileron has led to the selection of several unique material combinations and easily manufactured structural configurations. The advanced composite aileron is a direct replacement for the metal aileron with a weight savings of 23 percent.

30. Griffin, C. F.; Fogg, L. D.; Stone, R. L.; and Dunning, E. G.: *Advanced Composite Aileron for L-1011 Transport Aircraft, Task 1.* NASA CR-145370, July 1978.

81N19232

Structural design and maintainability criteria were established and used as a guideline for evaluating a variety of configurations and materials for each of the major subcomponents. From this array of subcomponent designs, several aileron assemblies were formulated and analyzed. The selected design is a multi-rib configuration with sheet skin covers mechanically fastened to channel-section ribs and spars. Thornel 300/5208 unidirectional tape was selected for the front spar and covers, and Thornel 300 fabric/5208 was chosen for the ribs.

31.   Jackson, A. C.; Crocker, J. F.; Ekvall, J. C.; Eudaily, R. R.; Mosesian, B.; Van Cleave, R. R.; and Vanhamersveld, J.: *Advanced Manufacturing Development of a Composite Empennage Component for L-1011 Aircraft. Phase 2: Design and Analysis.* NASA CR-165634, Apr. 1981.

83N30404#

The composite fin design consists of two one-piece cocured covers, two one-piece cocured spars, and eleven ribs. The lower ribs are truss ribs with graphite/epoxy caps and aluminum truss members. The upper three ribs are a sandwich design with graphite/epoxy face sheets and a syntactic epoxy core. The design achieves a 27% weight savings compared to the metal box. The fastener count has been reduced from over 40,000 to less than 7000. The structural integrity of the composite fin was verified by analysis and test. The static, fail-safe, and flutter analyses were completed. An extensive test program has established the material behavior under a range of conditions and critical subcomponents were tested to verify the structural concepts.

32.   Jackson, A. C.; and Dorwald, F.: *Advanced Manufacturing Development of a Composite Empennage Component for L-1011 Aircraft. Phase 4: Full Scale Ground Test.* NASA CR-166015, Dec. 1982.

83N14168#

The ground tests conducted on the advanced composite vertical fin (ACVF) are described. The design and fabrication of the test fixture and the transition structure, static test of Ground Test Article (GTA) No. 1, rework of GTA No. 2, and static, damage tolerance, fail-safe, and residual strength tests of GTA No. 2 are described.

33.   Jackson, A. C.: *Advanced Composite Vertical Fin for L-1011 Aircraft: Design, Manufacture and Test.* NASA CR-3816, Sept. 1984.

84X10452#

The structural box of the L-1011 vertical fin was redesigned using advanced composite materials. The box was fabricated and ground tested to verify the structural integrity. The complete program starting with the design and analysis and proceeding through the process development, ancillary test program, production readiness verification testing, fabrication of the full-scale fin boxes, and the full-scale ground testing is summarized. The program showed that advanced composites can economically and effectively be used in the design and fabrication of medium primary structures for commercial aircraft.

Static strength variability was demonstrated to be comparable to metal structures, and the long term durability of advanced composite components was demonstrated.

34.   Jackson, A. C.; Sandifer, J.; Sandorff, P.; and Van Cleave, R.: *Advanced Manufacturing Development of a Composite Empennage Component for L-1011 Aircraft. Phase III: Final Report—Production Readiness Verification Testing.* NASA CR-172383, Aug. 1984.

84X10452#

The Production Readiness Verification Tests (PRVT) were designed to provide information to answer the following questions: • What is the range of production qualities that can be expected for components manufactured under conditions similar to those expected in production, and how realistic and effective are proposed quality levels and quality control procedures? • What variability in static strength can be expected for production quality components, and are the margins sufficient to account for this variability? • Will production quality components survive extended time laboratory fatigue tests involving both load and environment simulation of sufficient duration and severity to provide confidence in in-service durability?

To provide data, 22 components of each of two key structural elements of the ACVF were fabricated for test. One element represented the front spar/fuselage attachment area, and the other element represented the cover/fuselage joint area. Ten of each element were static strength tested. The remainder were durability tested for up to the equivalent of 20 years of service. Reproducibility was demonstrated and the quality control procedures verified.

Static strength variability has been demonstrated to be comparable to metallic structures. The range of production qualities has been established. The long-term durability of advanced composite components has been demonstrated.

35.   James, A. M.; and Bohon, H. L.: Production Readiness Verification Testing. *Advances in Composite Materials* (Proceedings of the Third International Conference on Composite Materials, Paris, France, Aug. 26–29), *Volume 2,* Pergamon Press, 1980, pp. 1059–1074.

81A40569

A Production Readiness Verification Testing (PRVT) program has been established to determine if structures fabricated from advanced composites

can be committed on a production basis to commercial airline service. The program utilizes subcomponents which reflect the variabilities in structure that can realistically be expected from current production and quality control technology to estimate the production qualities, variation in static strength, and durability of advanced composite structures. The results of the static tests and a durability assessment after one year of continuous load/environment testing of twenty-two duplicates of each of two structural components (a segment of the front spar and cover of a vertical stabilizer box structure) are discussed.

36. Lehman, G. H.; Purdy, D. M.; Cominsky, A.; Hawley, A. V.; Amason, M. P.; Kung, J. T.; Palmer, R. J.; Purves, N. B.; Marra, P. J.; and Hancock, G. R.: *Advanced Composite Rudders for DC-10 Aircraft: Design, Manufacturing, and Ground Tests.* NASA CR-145068, Apr. 1976.

Design synthesis, tooling and process development, manufacturing, and ground testing of a graphite epoxy rudder for the DC-10 commercial transport are discussed. The composite structure was fabricated using a unique processing method in which the thermal expansion characteristics of rubber tooling mandrels were used to generate curing pressures during an oven cure cycle. The ground test program resulted in certification of the rudder for passenger-carrying flights. Results of the structural and environmental tests are interpreted and detailed development of the rubber tooling and manufacturing process is described.

37. Marra, P. J.: Trapped Rubber Processing for Advanced Composites. SME Paper EM76-172, 1976.

Trapped rubber processing is a molding technique for composites in which precast silicone rubber is placed within a closed cavity where it thermally expands against the composite's surface supported by the vessel walls. The method has been applied by the Douglas Aircraft Company, under contract to NASA Langley, to the design and fabrication of 10 DC-10 graphite/epoxy upper aft rudder assemblies. A three-bay development tool form mold die has been designed and manufactured, and tooling parameters have been established. Fabrication procedures include graphite layup, assembly of details in the tool, and a cure cycle. The technique has made it possible for the cocured fabrication of complex primary box structures otherwise impracticable via standard composite material processes.

38. Palmer, J. M.; Stephens, C. O.; and Sutton, J. O.: *DC-10 Composite Vertical Stabilizer Ground Test Program.* NASA CR-3715, Aug. 1983.

A review of the structural configuration and ground test program is presented. Particular emphasis is placed on the testing of a full-scale stub box test subcomponent and full-span ground test unit. The stub box subcomponent was tested in an environmental chamber under ambient, cold/wet, and hot/wet conditions. The test program included design limit static loads, fatigue spectrum loading to approximately two service lifetimes (with and without damage), design limit damage tolerance tests, and a final residual strength test to a structural failure. The first full-scale ground test unit was tested under ambient conditions. The test unit was to have undergone static, fatigue, and damage tolerance tests but a premature structural failure occurred at design limit load during the third limit load test. A failure theory was developed which explains the similarity in types of failure and the large load discrepancy at failure between the two test articles. The theory attributes both failures to high stress concentrations at the edge of the lower rear spar access opening. A second full-scale ground test unit has been modified to incorporate the various changes resulting from the premature failure. The article has been assembled and is active in the test program.

39. Soovere, J.: Sonic Fatigue Testing of an Advanced Composite Aileron. *Journal of Aircraft,* vol. 19, Apr. 1982, pp. 304–310. (Available as AIAA Paper 81-0634.)

The sonic fatigue test program to verify the design of the composite inboard aileron for the L-1011 airplane is described. The program covers the development of random fatigue data by means of coupon testing and modal studies on a representative section of the composite aileron, culminating in the accelerated sonic fatigue proof test. The composite aileron sustained nonlinear panel vibration during the proof test without failure. Viscous damping coefficients as low as 0.4% were measured on the panels. The effects of moisture conditioning and elevated temperature on the random fatigue life of both undamaged and impact damaged coupons were investigated. The combination of impact damage, moisture, and a 180°F temperature could reduce the random fatigue life by 50%.

40. Stephens, C. O.: *Advanced Composite Vertical Stabilizer for DC-10 Transport Aircraft.* NASA CR-173985, July 1978.

84N34573#

The structural design configuration for the composite vertical stabilizer is described and the structural design, analysis, and weight activities are presented. The status of fabrication and test activities for the development test portion of the program is described. Test results are presented for the skin panels, spar web, spar cap to cover, and laminate property specimens. Engineering drawings of verification test panels and root fittings, rudder support specimens, titanium fittings, and rear spar specimen analysis models are included.

ACEE wing and fuselage technology

41. Bunin, B. L.; and Sagui, R. L.: *Critical Joints in Large Composite Primary Aircraft Structures. Volume 3: Ancillary Test Results.* NASA CR-172588, June 1985.

86X10035#

A program was conducted to develop the technology for critical structural joints for composite wing structure that meets all the design requirements of a 1990 commercial transport aircraft. Results are given for a comprehensive ancillary test program consisting of single-bolt composite joint specimens tested in a variety of configurations. These tests were conducted to characterize the strength and load deflection properties that are required for multirow joint analysis.

42. Bunin, B. L.: *Critical Joints in Large Composite Primary Aircraft Structures. Volume 2: Technology Demonstration Test Report.* NASA CR-172587, June 1985.

86X10034#

A program was conducted to develop the technology for critical structural joints in composite wing structure that meets all the design requirements of a 1990 commercial transport aircraft. The results of four large composite multirow bolted joint tests are presented. The tests were conducted to demonstrate the technology for critical joints in highly loaded composite structure and to verify the analytical methods that were developed throughout the program. Discussions are given of the test article, instrumentation, test setup, test procedures, and test results for each of the four specimens. Some of the analytical predictions are also included.

43. Bunin, B. L.: *Critical Joints in Large Composite Primary Aircraft Structures. Volume 1: Technical Summary.* NASA CR-3914, Sept. 1985.

85X10405#

A program was conducted at Douglas Aircraft Company to develop the technology for critical joints in composite wing structure that meets all the design requirements of a 1990 commercial transport aircraft. The analysis methodology development, structural test program, and correlation between test results and analytical strength predictions are reviewed.

44. Bunin, B. L.: *Critical Composite Joint Subcomponents: Analysis and Test Results.* NASA CR-3711, Sept. 1983.

83X10324#

A prime objective of the program was to demonstrate the ability to reliably predict the strength of large bolted composite joints. Load sharing between bolts in multirow joints was computed by a nonlinear analysis program (A4EJ) which was used both to assess the efficiency of different joint design concepts and to predict the strengths of large test articles representing a section from a wing root chordwise splice. In most cases, the predictions were accurate to within a few percent of the test results. A highlight of these tests was the consistent ability to achieve gross-section failure strains on the order of 0.005 which represents a considerable improvement over the state of the art. The improvement was attained largely as the result of the better understanding of the load sharing in multirow joints provided by the analysis. The typical load intensity on the structural joints was about 40 to 45 thousand pounds per inch.

45. Griffin, C. F.; and James A. M.: *Fuel Containment, Lightning Protection and Damage Tolerance in Large Composite Primary Aircraft Structures.* NASA CR-3875, May 1985.

85X10299#

The damage-tolerance characteristics of high strain-to-failure graphite fibers and toughened resins were evaluated. Test results show that conventional fuel tank sealing techniques are applicable to composite structures. Techniques were developed to prevent fuel leaks due to low-energy impact damage. For wing panels subjected to swept stroke lightning strikes, a surface protection of graphite/aluminum wire fabric and a fastener treatment proved effective in eliminating internal sparking and reducing structural damage. The technology features developed

were incorporated and demonstrated in a test panel designed to meet the strength, stiffness, and damage-tolerance requirements of a large commercial transport aircraft. The panel test results exceeded design requirements for all test conditions.

46. Griffin, C. F.: *Fuel Containment and Damage Tolerance in Large Composite Primary Aircraft Structures.* NASA CR-166083, Mar. 1983.

85N30032#

Technical problems related to fuel containment and damage tolerance of composite material wings for transport aircraft were investigated. The major tasks were the following: (1) the preliminary design of damage tolerant wing surface using composite materials; (2) the evaluation of fuel sealing and lightning protection methods for a composite material wing; and (3) an experimental investigation of the damage tolerance characteristics of toughened resin graphite/epoxy materials.

47. Harvey, S. T.; and Michaelson, G. L.: *Advanced Composites Wing Study Program, Volume 2.* NASA CR-145382, Vol. 2, Aug. 1978.

78N32186#

The study objective was to develop a plan to define the effort needed to support a production commitment for the extensive use of composite materials in wings of new generation aircraft that will enter service in the 1985-1990 time period. Identification and analysis of what was needed to meet the above plan requirements resulted in a program plan consisting of three key development areas: (1) technology development; (2) production capability development; and (3) integration and validation by designing, building, and testing major development hardware.

48. Harvey, S. T.; and Michaelson, G. L.: *Advanced Composites Wing Study Program. Volume 1: Executive Summary.* NASA CR-145382, Vol. 1, Sept. 1978.

78N32185#

The effort necessary to achieve a state of production readiness for the design and manufacturing of advanced composite wing structure is outlined. Technical assessment and program options are also reviewed for the wing study results.

49. Jackson, A. C.; Campion, M. C.; and Pei, G.: *Study of Utilization of Advanced Composites in Fuselage Structures of Large Transports.* NASA CR-172404, Sept. 1984.

84X10475#

The effort required by the transport aircraft manufacturers to support the introduction of advanced composite materials into the fuselage structure of future commercial and military transport aircraft is investigated. Technology issues, potential benefits to military life cycle costs and commercial operating costs, and development plans are examined. The most urgent technology issues defined are impact dynamics, acoustic transmission, pressure containment and damage tolerance, post-buckling, cutouts, and joints and splices. A technology demonstration program is defined.

50. Johnson, R. W.; Thomson, L. W.; and Wilson, R. D.: *Study on Utilization of Advanced Composites in Fuselage Structures of Large Transports.* NASA CR-172406, Feb. 1985.

85X10226#

The potential for utilizing advanced composites in fuselage structures of large transports was studied. Six fuselage design concepts were selected and evaluated in terms of structural performance, weight, and manufacturing development and costs. Two concepts were selected that merit further consideration for composite fuselage application. These concepts are (1) a full-depth honeycomb design with no stringers, and (2) an I-section stringer stiffened laminate skin design. Weight reductions due to applying composites to the fuselages of commercial and military transports were calculated. The benefits of applying composites to a fleet of military transports were determined. Significant technology issues pertinent to composite fuselage structures were identified and evaluated.

51. McCarty, J. E.; and Roeseler, W. G.: *Durability and Damage Tolerance of Large Composite Primary Aircraft Structure (LCPAS).* NASA CR-3767, Jan. 1984.

84X10135#

Analysis and testing that addressed the key technology areas of durability and damage tolerance were completed for wing surface panels. The wing of a fuel-efficient, 200-passenger commercial transport airplane for 1990 delivery was sized using graphite-epoxy materials. Coupons of various layups used in the wing sizing were tested in tension, compression, and spectrum fatigue with typical fastener penetrations. The compression strength after barely visible impact damage was determined from coupon and structural element tests. The results of the coupon and element tests were used to design three distinctly different compression panels meeting the strength,

stiffness, and damage-tolerance requirements of the upper wing panels. These three concepts were tested with various amounts of damage ranging from barely visible impact to through-penetration.

52.   Nelson, W. D.; Bunin, B. L.; and Hart-Smith, L. J.: *Critical Joints in Large Composite Aircraft Structure.* NASA CR-3710, Aug. 1983.

83X10287#

A program was conducted at Douglas Aircraft Company under NASA-Langley Contract NAS1-16857 to develop the technology for critical structural joints of composite wing structure that meets design requirements for a 1990 commercial transport aircraft. The prime objective of the program was to demonstrate the ability to reliably predict the strength of large bolted composite joints. Ancillary testing of 180 specimens generated data on strength and load deflection characteristics which provided input to the joint analysis. Load-sharing between fasteners in multirow bolted joints was computed by the nonlinear analysis program A4EJ. This program was used to predict strengths of 20 additional large subcomponents representing strips from a wing root chordwise splice. In most cases, the predictions were accurate to within a few percent of the test results. In some cases, the observed mode of failure was different than anticipated. The highlight of subcomponent testing was the consistent ability to achieve gross-section failure strains close to 0.005.

53.   Sakata, I. F.; and Ostrom, R. B.: *Study on Utilization of Advanced Composites in Commercial Aircraft Wing Structures, Volume 2.* NASA CR-145381, Vol. 2, Aug. 1978.

78N32188#

A plan is defined for a composite wing development effort which will assist commercial transport manufacturers in reaching a level of technology readiness where the utilization of composite wing structure is a cost competitive option for a new aircraft. The recommended development effort consists of two programs: a joint government/industry material development program and a wing structure development program. Both programs are described in detail.

54.   Sakata, I. F.; Ostrom, R. B.; and Cardinale, S. V.: *Study on Utilization of Advanced Composites in Commercial Aircraft Wing Structures. Volume 1: Executive Summary.* NASA CR-145381, Vol. 1, Aug. 1978.

78N32187#

The effort required by commercial transport manufacturers to accomplish the transition from current construction materials and practices to extensive use of composites in aircraft wings was investigated. A conceptual design of an advanced technology reduced energy aircraft provided the framework for identifying and investigating unique design aspects. A plan defines the essential technology needs and formulates approaches for effecting the required wing development. The wing development program plans, resource needs, and recommendations are summarized.

55.   Sandifer, J. P.: *Fuel Containment and Damage Tolerance for Large Composite Primary Aircraft Structures. Phase 1: Testing.* NASA CR-166091, Mar. 1983.

85N35241#

Technical problems associated with fuel containment and damage tolerance of composite material wings for transport aircraft were identified. The major tasks are the following: (1) the preliminary design of damage tolerant wing surface using composite materials; (2) the evaluation of fuel sealing and lightning protection methods of a composite material wing; and (3) an experimental investigation of the damage tolerance characteristics of toughened resin graphite/epoxy materials. The test results, the test techniques, and the test data are presented.

56.   Sandifer, J. P.; Denny, A.; and Wood, M. A.: *Fuel Containment and Damage Tolerance in Large Composite Primary Aircraft Structures. Phase 2: Testing.* NASA CR-172519, Apr. 1985.

85X10189#

Technical issues associated with fuel containment and damage tolerance of composite wing structures for transport aircraft were investigated. A test series was conducted on graphite/epoxy box beams simulating a wing cover to spar cap joint configuration of a pressurized fuel tank. These tests evaluated the effectiveness of sealing methods with various fastener types and spacings under fatigue loading and with pressurized fuel. Another test series evaluated the ability of selected coatings, films, and materials to prevent fuel leakage through 32-ply AS4/2220-1 laminates at various impact energy levels. Compression tests were also performed on panels subjected to Zone 2 lightning strikes. All of these data were integrated into a demonstration article representing a moderately loaded area of a transport wing.

57.   Smith, P. J.; Wilson, R. D.; and Gibbins, M. N.: *Damage Tolerant Composite Wing Panels for Transport Aircraft.* NASA CR-3951, Dec. 1985.

86X10044

Commercial aircraft advanced composite wing surface panels were tested for durability and damage tolerance. The wing of a fuel-efficient, 200-passenger airplane for 1990 delivery was sized using grahite-epoxy materials. The damage tolerance program was structured to allow a systematic progression from material evaluations to the optimized large panel verification tests. The program included coupon testing to evaluate toughened material systems, static and fatigue tests of compression coupons with varying amounts of impact damage, and element tests of three-stiffener panels to evaluate upper wing panel design concepts. The wing structure damage environment was studied. A series of technology demonstration tests of large compression panels were performed. A repair investigation was included in the final large panel test.

58. Watts, D. J.; Sumida, P. T.; Bunin, B. L.; Janicki, G. C.; Walker, J. V.; and Fox, B. R.: *A Study of the Utilization of Advanced Composites in Fuselage Structures of Commercial Aircraft.* NASA CR-172405, Jan. 1985.

85X10225#

A conceptual composite fuselage was designed, retaining the basic MD-100 structural arrangement for doors, windows, wing, wheel wells, cockpit enclosure, major bulkheads, and interfaces with existing aircraft systems and cabin interior arrangements. A 32-percent weight savings from the existing MD-100 design was realized for this design.

59. Watts, D. J.: *A Study on the Utilization of Advanced Composites in Commercial Aircraft Wing Structure.* NASA CR-158902, Vol. 2, July 1978.

79N20190#

A study was conducted to define the technology and data needed to support the introduction of advanced composite materials in the wing structure of future production aircraft.

60. Watts, D. J.: *A Study on the Utilization of Advanced Composites in Commercial Aircraft Wing Structure: Executive Summary.* NASA CR-158902, Vol. 1, July 1978.

79N20189#

The overall wing study objectives were to study and plan the effort by commercial transport aircraft manufacturers to accomplish the transition from current conventional materials and practices to extensive use of advanced composites in wings of aircraft that will enter service in the 1985–1990 time period. Specific wing study objectives were to define the technology and data needed to support an aircraft manufacturer's commitment to utilize composite primary wing structures in future production aircraft and to develop plans for a composite wing technology program which will provide the needed technology and data.

Langley Basic Technology

Design and analysis

61. Adelman, H. M.: *Preliminary Design Procedure for Insulated Structures Subjected to Transient Heating.* NASA TP-1534, Dec. 1979.

80N12438#

Minimum-mass designs were obtained for insulated structural panels loaded by a general set of inplane forces and a time dependent temperature. Temperature and stress histories in the structure are given by closed-form solutions, and optimization of the insulation and structural thicknesses is performed by nonlinear mathematical programming techniques. Design calculations are described to evaluate the structural efficiency of eight materials under combined heating and mechanical loads: graphite/polyimide, graphite/epoxy, boron/aluminum, titanium, aluminum, Rene 41, carbon/carbon, and Lockalloy. The effects on design mass of intensity and duration of heating were assessed. Results indicate that an optimum structure may have a temperature response well below the recommended allowable temperature for the material.

62. Agarwal, B.; and Davis, R. C.: *Minimum-Weight Designs for Hat-Stiffened Composite Panels Under Uniaxial Compression.* NASA TN D-7779, Nov. 1977.

75N10497#

Optimum hat-stiffened compression panel designs are determined by a structural synthesis technique. The effects of simplifying assumptions made in the buckling analysis for the optimization program are investigated by a linked plate element program. Optimization results for hat-stiffened graphite-epoxy panels show a 50-percent weight savings over optimized aluminum panels. Composite panels are shown to possess a variety of proportions at nearly constant weight.

63. Anderson, M. S.; Stroud, W. J.; Durling, B. J.; and Hennessy, K. W.: *PASCO: Structural Panel Analysis and Sizing Code, Users Manual—Revised.* NASA TM-80182, Nov. 1981.

82N20563

A computer code denoted PASCO is described for analyzing and sizing uniaxially stiffened composite panels. Buckling and vibration analyses are carried out with a linked plate analysis computer code denoted VIPASA, which is included in PASCO. Sizing is based on nonlinear mathematical programming techniques and employs a computer code denoted CONMIN, also included in PASCO. Design requirements considered are initial buckling, material strength, stiffness, and vibration frequency. A user's manual for PASCO is presented.

64. Anderson, M. S.; and Stroud, W. J.: A General Panel Sizing Computer Code and Its Application to Composite Structural Panels. *19th AIAA Structures, Structural Dynamics and Materials Conference* (Bethesda, Md., Apr. 3–5), *Technical Papers*, 1978, pp. 14–22. (Available as AIAA Paper 78-467.)

78A29778#

A computer code for obtaining the dimensions of optimum (least mass) stiffened composite structural panels is described. The procedure, which is based on nonlinear mathematical programming and a rigorous buckling analysis, is applicable to general cross sections under general loading conditions causing buckling. A simplified method of accounting for bow-type imperfections is also included. Design studies in the form of structural efficiency charts for axial compression loading are made with the code for blade and hat stiffened panels. The effects on panel mass of imperfections, material strength limitations, and panel stiffness requirements are also examined. Comparisons with previously published experimental data show that accounting for imperfections improves correlation between theory and experiment.

65. Arnold, R. R.; and Parekh, J. C.: *Theoretical Prediction of Ultimate Strength of Composite Curved Frame Members.* NASA CR-172441, Oct. 1984.

84X10525#

The research presented herein concerns the development of a theoretical approach for the prediction of ultimate strength of composite curved frame members. The theoretical derivation includes important physically observable characteristics of composites not usually included in conventional elasticity theories, such as effects of transverse shear, nonlinear geometry, and nonlinear material behavior. The

formulation utilizes an energy approach based on the Reissner variational principle in which both displacement and stress are taken to be variationally unknown.

66. Barthelemy, J.-F. M.; and Sobieszczanski-Sobieski, J.: Extrapolation of Optimum Design Based on Sensitivity Derivatives. *AIAA Journal,* vol. 21, May 1983, pp. 797–799.

83A28978#

In many applications, it is useful to have information regarding the effect of the perturbations of the design problem constants (parameters) on the optimum solution of the problem. The present investigation is concerned with an example of optimum sensitivity analysis involving a composite panel. The case was selected in connection with the nonlinearity of the constraints and the tendency for a set of active constraints to change when the parameters are disturbed.

67. Bauld, N. R., Jr.; and Goree, J. G.: *An Investigation of the Accuracy of Finite Difference Methods in the Solution of Linear Elasticity Problems.* NASA CR-173015, Aug. 1983.

83N34348#

The accuracy of the finite difference method in the solution of linear elasticity problems that involve either a stress discontinuity or a stress singularity is considered. Solutions to three elasticity problems are discussed in detail: a semi-infinite plane subjected to a uniform load over a portion of its boundary; a bimetallic plate under uniform tensile stress; and a long, midplane symmetric, fiber reinforced laminate subjected to uniform axial strain. Finite difference solutions to the three problems are compared with finite element solutions to corresponding problems. For the first problem a comparison with the exact solution is also made.

68. Biggers, S. B.; and Dickson, J. N.: *POSTOP: Postbuckled Open-Stiffener Optimum Panels, User's Manual.* NASA CR-172260, Jan. 1984.

84N18682#

The computer program POSTOP, developed to serve as an aid in the analysis and sizing of stiffened composite panels that may be loaded in the postbuckling regime, is intended for the preliminary design of metal or composite panels with open-section stiffeners, subjected to multiple combined biaxial compression (or tension), shear, and normal pressure load cases. Longitudinal compression, however, is

assumed to be the dominant loading. Temperature, initial bow eccentricity, and load eccentricity effects are included.

69. Boitnott, R. L.; Johnson, E. R.; and Starnes, J. H., Jr.: A Nonlinear Analysis of Infinitely Long Graphite-Epoxy Cylindrical Panels Loaded With Internal Pressure. *26th AIAA Structures, Structural Dynamics and Materials Conference* (Orlando, Fla., Apr. 15–17), *Technical Papers, Part 1,* 1985, pp. 593–604. (Available as AIAA Paper 85-0770.)

85A30291#

The structural response of internally pressurized composite cylindrical panels, representative of a transport aircraft's fuselage skins, is predicted by means of a one-dimensional, geometrically nonlinear analysis. An analytical study is conducted for the response of 4-, 8-, and 16-ply graphite/epoxy skins.

70. Boitnott, R. L.; Starnes, J. H., Jr.; and Johnson, E. R.: Nonlinear Response and Failure Characteristics of Clamped Internally Pressurized Graphite-Epoxy Cylindrical Panels. *25th AIAA Structures, Structural Dynamics and Materials Conference* (Palm Springs, Calif., May 14–16), *Technical Papers, Part 1,* 1984, pp. 514–525. (Available as AIAA Paper 84-0955.)

84A31680#

The nonlinear response and failure characteristics of internally pressurized 4- to 16-ply cylindrical panels of graphite/epoxy are experimentally and analytically studied. Tests simulated the skin of a typical transport aircraft fuselage under loading to eventual failure. Aluminum test specimens were similarly tested and compared. All graphite/epoxy panels are found to withstand the internal pressures applied, which are well above the proof pressure loading expected for transport fuselage structures.

71. Byers, B. A.; and Stoecklin, R. L.: *Preliminary Design of Graphite Composite Wing Panels for Commercial Transport Aircraft.* NASA CR-159150, Feb. 1980.

80N17148#

Subjectively assessed practical and producible graphite/epoxy designs were subjected to a multilevel screening procedure which considered structural functions, efficiency, manufacturing and producibility, costs, maintainability, and inspectability. The selected panel design showed a weight savings of 25 percent over a conventional aluminum design meeting the same design requirements. The estimated cost

reduction in manufacturing was 20 percent, based on 200 aircraft and projected 1985 automated composites manufacturing capability.

72. Carper, D. M.; Johnson, E. R.; and Hyer, M. W.: *The Response of Cylindrical Panels Fabricated From Symmetrically and Unsymmetrically Laminated Composite Materials.* NASA CR-170178, Apr. 1983.

83N22327

Equations are developed which govern the deflection response of long cylindrical panels subjected to a line load. The line load is directed toward the center of curvature of the panel, is located at an arbitrary point along the arc length of the panel, and is included at an arbitrary angle relative to the radial direction. Both symmetrically laminated and the less common unsymmetrically laminated simply supported panels are studied.

73. Carper, D. M.; Hyer, M. W.; and Johnson, E. R.: *Large Deformation Behavior of Long Shallow Cylindrical Composite Panels.* NASA CR-173286, Sept. 1983.

84N22975#

An exact solution is presented for the large deformation response of a simply supported orthotropic cylindrical panel subjected to a uniform line load along a cylinder generator. The cross section of the cylinder is circular and deformations up to the fully snapped through position are investigated. Experimental results of displacement controlled tests performed on graphite-epoxy curved panels are compared with analytic predictions. Results demonstrate that panel shallowness, material orthotropy, and stacking sequence can influence the nonlinear static response. Initial geometric imperfections, observed during testing, were found to influence the response of the panels. However, the overall correlation of analytic and experimental results was good.

74. Chen, J. K.; and Sun, C. T.: Nonlinear Transient Responses of Initially Stressed Composite Plates. *Computers and Structures,* vol. 21, no. 3, 1985, pp. 513–520.

85A44743

The nonlinear transient response of initially stressed composite plates is investigated using the finite element method. A nine-node isoparametric quadrilateral element was developed to model laminated plates under initial deformation and initial stress according to the Mindlin plate theory and

von Karman large deflection assumptions. In the time integration, the Newmark constant acceleration method in conjunction with an efficient and accurate iteration scheme is used. Numerical results for deflections and bending moments for isotropic and laminated plates are obtained.

75. Cohen, G. A.: FASOR—A Second Generation Shell of Revolution Code. *Trends in Computerized Structural Analysis and Synthesis* (Proceedings of Symposium, Washington, D.C., Oct. 30–Nov. 1), Pergamon Press, 1978, pp. 301–309.

79A22948

An integrated computer program entitled Field Analysis of Shells of Revolution (FASOR) currently under development for NASA is described. When completed, this code will treat prebuckling, buckling, initial postbuckling, and vibrations under axisymmetric static loads as well as linear response and bifurcation under asymmetric static loads. Although these modes of response are treated by existing programs, FASOR extends the class of problems treated to include general anisotropy and transverse shear deformations of stiffened laminated shells.

76. Cohen, G. A.: Transverse Shear Stiffness of Laminated Anisotropic Shells. *Computer Methods in Applied Mechanics and Engineering,* vol. 13, Feb. 1978, pp. 205–220. (Presented at 15th Midwestern Mechanics Conference, University of Illinois, Chicago, Ill., Mar. 23–25, 1977.)

78A30894

Equations are derived for the transverse shear stiffness of laminated anisotropic shells. The equations are based on Taylor series expansions about a generic point for stress resultants and couples, identically satisfying plate equilibrium equations. These equations are used to find statically correct expressions for in-surface stresses, transverse shear stresses, and the area density of transverse shear strain energy, in terms of transverse shear stress resultants and redundants.

77. Davis, G. W.; and Sakata, I. F.: *Design Considerations for Composite Fuselage Structure of Commercial Transport Aircraft.* NASA CR-159296, Mar. 1981.

81N22419#

The structural, manufacturing, and service and environmental considerations that could impact the design of composite fuselage structure for commercial transport aircraft application were explored. The severity of these considerations was assessed and the principal design drivers delineated. Technical issues and potential problem areas which must be resolved before sufficient confidence is established to commit to composite materials were defined. The key issues considered are: definition of composite fuselage design specifications, damage tolerance, and crashworthiness.

78. Dickson, J. N.; and Biggers, S. B.: *POSTOP: Postbuckled Open-Stiffener Optimum Panels—Theory and Capability.* NASA CR-172259, Jan. 1984.

84N18679#

The computer program POSTOP was developed to serve as an aid in the analysis and sizing of stiffened composite panels that are loaded in the postbuckling regime. A comprehensive set of analysis routines was coupled to a widely used optimization program to produce this sizing code. POSTOP is intended for the preliminary design of metal or composite panels with open-section stiffeners, subjected to multiple combined biaxial compression (or tension), shear, and normal pressure load cases. Longitudinal compression, however, is assumed to be the dominant loading. Temperature, initial bow eccentricity, and load eccentricity effects are included.

79. Dickson, J. N.; and Biggers, S. B.: *Design and Analysis of a Stiffened Composite Fuselage Panel.* NASA CR-159302, Aug. 1980.

80N31820#

The design and analysis of a stiffened composite panel that is representative of the fuselage structure of existing wide bodied aircraft are discussed. The panel is a minimum weight design, based on the current level of technology and realistic loads and criteria. Several different stiffener configurations were investigated in the optimization process. The final configuration is an all graphite/epoxy J-stiffened design in which the skin between adjacent stiffeners is permitted to buckle under design loads. Fail safe concepts typically employed in metallic fuselage structure have been incorporated in the design. A conservative approach has been used with regard to structural details such as skin/frame and stringer/frame attachments and other areas where sufficient design data were not available.

80. Douglas, D. O.; Holzmacher, D. E.; Lane, Z. C.; and Thornton, E. A.: *Studies of Finite Element Analysis of Composite Material Structures.* NASA CR-119144, Sept. 1975.

75N32508#

Research in the area of finite element analysis is summarized. Topics discussed include finite element analysis of a picture frame shear test, BANSAP (a bandwidth reduction program for SAP IV), FEMESH (a finite element mesh generation program based on isoparametric zones), and finite element analysis of a composite bolted joint specimen.

81. Eckstrom, C. V.; and Spain, C. V.: *Design Considerations and Experiences in the Use of Composite Material for an Aeroelastic Research Wing. 23rd AIAA Structures, Structural Dynamics and Materials Conference* (New Orleans, La., May 10–12), *Technical Papers, Part 2*, 1982, pp. 157–165. (Available as AIAA Paper 82-0678.)

82A30146#

Experiences in using composite skin material on an aeroelastic research wing used in flight flutter testing are described. Significant variations in skin shear modulus due to stress and temperature were encountered with the original fiberglass laminate skin designed to minimize wing torsional stiffness. These variations along with the sensitivity of wing torsional stiffness to the skin-to-frame attachment method complicated the structural model vibration mode predictions. A wing skin redesign with different fiber orientation and a reduction in the amount of skin-to-frame bonding resulted in more predictable modal characteristics without sacrificing design objectives. Design and modeling considerations for future applications are discussed.

82. Erb, D. A.; Cooper, P. A.; and Weisshaar, T. A.: *An Exact Plane-Stress Solution for a Class of Problems in Orthotropic Elasticity.* NASA TM-84736, Feb. 1982.

82N23552#

An exact solution for the stress field within a rectangular slab of orthotropic material is found using a two dimensional Fourier series formulation. The material is required to be in plane stress, with general stress boundary conditions, and the principle axes of the material must be parallel to the sides of the rectangle. Two load cases similar to those encountered in materials testing are investigated using the solution. The solution method has potential uses in stress analysis of composite structures.

83. Farley, G. L.; and Baker, D. J.: *Graphics and Composite Material Computer Program Enhancements for SPAR.* NASA TM-80209, Feb. 1980.

80N18750#

User documentation is provided for computer programs developed for use in conjunction with SPAR. These programs plot digital data, simplify input for composite material section properties, and compute lamina stresses and strains. Sample problems are presented including execution procedures, program input, and graphical output.

84. Fichter, W. B.: *Stress Decay in an Orthotropic Half-Plane Under Self-Equilibrating Sinusoidal Loading.* NASA TM-85791, May 1984.

84N27066

An elastic orthotropic half-plane subjected to sinusoidal normal loading along an entire straight edge is analyzed. Stresses are calculated for material property combinations which are representative of some fiber reinforced composites. Plots of the stresses as functions of the distance from the loaded boundary show that they can differ greatly from their counterparts in the isotropic half-plane under the same loading. How the results impact the question of the applicability of St. Venant's principle to orthotropic materials is briefly discussed.

85. Fleury, C.; and Schmit, L. A., Jr.: *Dual Methods and Approximation Concepts in Structural Synthesis.* NASA CR-3226, Dec. 1980.

81N14347#

Approximation concepts and dual method algorithms are combined to create a method for minimum weight design of structural systems. Approximation concepts convert the basic mathematical programming statement of the structural synthesis problem into a sequence of explicit primal problems of separable form. These problems are solved by constructing explicit dual functions, which are maximized subject to nonnegativity constraints on the dual variables.

86. Gray, C. E., Jr.; Mei, C.; and Decha-Umphai, K.: Large Deflection, Large Amplitude Vibrations and Random Response of Symmetrically Laminated Rectangular Plates. *25th AIAA Structures, Structural Dynamics and Materials Conference* (Palm Springs, Calif., May 14–16), *Technical Papers, Part 1*, 1984, pp. 452–463. (Available as AIAA Paper 84-0909.)

84A31674#

An analytical method is presented for determining large-deflection static bending, large-amplitude free and forced vibrations, and large-amplitude random response of a clamped, symmetrically laminated, rectangular, thin plate subjected to a uniformly distributed transverse loading. Both movable and immovable inplane boundary conditions are

considered. Numerical results for bending deflections and strains, frequency ratios, mean-square center deflections, and mean-square maximum strains are presented showing the parametric effects of plate length-to-width ratio, orientation of layers, and intensities of applied force for both the linear and nonlinear cases. The analytical results for large-deflection random response are verified through comparison with experimental data.

87. Griffin, O. H., Jr.; and Kamat, M. P.: *Three Dimensional Inelastic Finite Element Analysis of Laminated Composites.* NASA CR-163712, Nov. 1980.

81N11114#

Formulations of the inelastic response of laminated composites to thermal and mechanical loading are used as the basis for development of the NALCOM (Nonlinear Analysis of Laminated Composites) computer program which uses a fully three dimensional isoparametric finite element with 24 nodes and 72 degrees of freedom. Elastic and elastic-plastic responses of boron/epoxy and graphite/epoxy are analyzed.

88. Gurdal, Z.; and Haftka, R. T.: Design of Stiffened Composite Panels With a Fracture Constraint. *Computers and Structures,* vol. 20, no. 1–3, 1985, pp. 457–465. (Presented at the Symposium on Advances and Trends in Structures and Dynamics, Washington, D.C., Oct. 22–25, 1984.)

85A41129

An automated procedure for designing minimum-weight composite panels subject to a local damage constraint under tensile loading was developed. A finite element program based on linear elastic fracture mechanics for calculating stress intensity factors (SIF) was incorporated in the design cycle. Panel fracture toughness was obtained by using a strain based criterion. A general purpose mathematical optimization algorithm was used for the weight minimization. Analytical sensitivity derivatives of the SIF employing the adjoint variable technique were used to enhance the computational efficiency of the procedure. Design results for both unstiffened and stiffened plates are presented.

89. Haftka, R. T.: Structural Optimization With Aeroelastic Constraints—A Survey of US Applications. *International Symposium on Aeroelasticity* (Nuremberg, West Germany, Oct. 5–7, 1981), *Collected Papers,* Deutsche Gesellschaft fuer Luft- und Raumfahrt, Cologne, 1982, pp. 179–186.

83A49188

The paper describes recent developments in the United States in the application of structural optimization techniques to problems of design under aeroelastic constraints. The material is divided into sections on (1) conventional strength design using aeroelastically calculated loads; (2) aeroelastic tailoring for improved performance; (3) design under flutter and static aeroelastic constraints; and (4) miscellaneous applications. Because of the high cost of applying formal optimization techniques to practical design problems, there have been very few applications of structural optimizations to actual aircraft. The paper is focused on trends that may eventually reverse this situation.

90. Haftka, R. T.; and Starnes, J. H., Jr.: Use of Optimum Stiffness Tailoring To Improve the Compressive Strength of Composite Plates With Holes. *26th AIAA Structures, Structural Dynamics and Materials Conference* (Orlando, Fla., Apr. 15–17), *Technical Papers, Part 1,* 1985, pp. 425–431. (Available as AIAA Paper 85-0721.)

85A30275#

A structural optimization procedure is used to tailor the cross-sectional stiffness distribution of compression-loaded composite plates with holes. The plate interior region contains the hole and is designed from a softer material system with a higher failure strain than the plate exterior region. All-graphite-epoxy plates and hybrid graphite/glass-epoxy plates were studied. The results show that cross-sectional stiffness tailoring can increase the compressive strength and decrease the mass of compression-loaded laminated plates with holes.

91. Herakovich, C. T.: Composite Laminates With Negative Through-the-Thickness Poisson's Ratios. *Journal of Composite Materials,* vol. 18, Sept. 1984, pp. 447–455.

85A17743

A simple analysis using two-dimensional lamination theory combined with the appropriate three-dimensional anisotropic constitutive equation is presented to show some rather surprising results for the range of values of the through-the-thickness effective Poisson's ratio for angle ply laminates. Results for graphite-epoxy show that the through-the-thickness effective Poisson's ratio can range from a high of 0.49 to a low of −0.21.

92. Hertz, T. J.; Shirk, M. H.; Ricketts, R. H.; and Weisshaar, T. A.: On the Track of Practical Forward-Swept Wings. *Astronautics and Aeronautics,* vol. 20, Jan. 1982, pp. 40–52.

82A19071

Structural laminates which comprise wing-cover skins for forward swept winged aircraft are examined. The laminates are themselves composed of lamina arranged in a symmetrical and unbalanced fashion. The fibers are oriented so that no fiber has a counterpart in the same ply which is at an exact anti-angle to itself. The laminate orientation creates a wash-out in a forward swept wing and alleviates aeroelastic loading. Further discussion is devoted to center-of-pressure movement, flutter behavior, aeroelasticity and aeroelastic divergence, and wind tunnel testing of aerodynamically tailored wings. It is found that rotating the laminate to increase the divergence dynamic pressure decreases strain under aerodynamic loading. Flight tests with three models are reported, and it is concluded that divergence can be avoided by the use of an efficient composite structure.

93. Housner, J. M.; and Stein, M.: *Numerical Analysis and Parametric Studies of the Buckling of Composite Orthotropic Compression and Shear Panels.* NASA TN D-7996, Oct. 1975.

75N33431#

A computer program is presented which was developed for the combined compression and shear of stiffened variable thickness orthotropic composite panels on discrete springs: boundary conditions are general and include elastic boundary restraints. Buckling solutions are obtained by using a newly developed trigonometric finite difference procedure which improves the solution convergence rate over conventional finite difference methods. The classical general shear buckling results, which exist only for simply supported panels over a limited range of orthotropic properties, were extended to the complete range of these properties for simply supported panels and, in addition, to the complete range of orthotropic properties for clamped panels. The program was also applied to parametric studies which examine the effect of filament orientation upon the buckling of graphite-epoxy panels. These studies included an examination of the filament orientations which yield maximum shear or compressive buckling strength for panels having all four edges simply supported or clamped over a wide range of aspect ratios.

94. Hyer, M. W.; Cooper, D. E.; and Cohen, D.: *Stresses and Deformations in Cross-Ply Composite Tubes Subjected to a Uniform Temperature Change: Elasticity and Approximate Solutions.* NASA CR-176147, June 1985.

85N33545#

The effects of a uniform temperature change on the stresses and deformations of composite tubes are investigated. The accuracy of an approximate solution based on the principle of complementary virtual work is determined. Interest centers on tube response away from the ends and so a planar elasticity approach is used. For the approximate solution, a piecewise linear variation of stresses with the radial coordinate is assumed. The results from the approximate solution are compared with the elasticity solution. The stress predictions agree well, particularly peak interlaminar stresses. Surprisingly, the axial deformations also agree well despite the fact that the deformations predicted by the approximate solution do not satisfy the interface displacement continuity conditions required by the elasticity solution.

95. Hyer, M. W.; and Cohen, D.: Residual Thermal Stresses in Cross-Ply Graphite-Epoxy Tubes. *1984 Advances in Aerospace Sciences and Engineering: Structures, Materials, Dynamics, and Space Station Propulsion* (Proceedings of Winter Annual Meeting, New Orleans, La., Dec. 9–14), ASME, 1984, pp. 87–93.

86A11337#

Elasticity solutions for the residual stress state in cross-ply graphite-epoxy tubes subjected to a spatially uniform temperature below the cure temperature are derived. The solutions are valid away from the ends of the tube. Numerical results are presented for three tubes. It is seen that stacking arrangement can affect the residual stresses and that the radial stresses are smaller than the other stress components. The results show that classical lamination theory is in considerable error when the tube radius to wall thickness ratio is less than 20.

96. Hyer, M. W.: Nonlinear Effects of Elastic Coupling in Unsymmetric Laminates. *Mechanics of Composite Materials: Recent Advances* (Proceedings of Symposium, Blacksburg, Va., Aug. 16–19, 1982), Pergamon Press, 1983, pp. 243–258.

84A33388

Classical lamination theory predicts the room-temperature shape of all unsymmetrically laminated, elevated-temperature cure composites to be a saddle shape. Experimental observation indicates, however, that in many cases the room-temperature shape is cylindrical. In addition, a second cylindrical shape can often be obtained from the first by a simple snap-through action. It is the elastic couplings between inplane and out-of-plane deformations which are inherent in unsymmetric laminates that are responsible for the room-temperature shape. However, the

couplings are so strong that geometrically nonlinear effects are produced. These effects are not accounted for in the classical theory. This paper reviews a theory developed to explain the effects of the coupling on laminate shape.

97. Hyer, M. W.: Unsymmetrically Laminated Composites. *Progress in Science and Engineering of Composites* (Proceedings of Fourth International Conference on Composite Materials, Tokyo, Japan, Oct. 25–28), *Volume 1,* Japan Society for Composite Materials/North-Holland, 1982, pp. 373–380.

83A40163

Classical lamination theory predicts that the room-temperature shapes of all elevated-temperature cure, unsymmetrically laminated composites are saddles. However, experimental observation indicates that the shapes are often cylindrical. In addition, a second cylindrical shape can sometimes be obtained from the first by a simple snap-through action. A geometrically nonlinear extension to classical lamination theory is used to explain this behavior. Approximate solutions to the nonlinear extension are obtained by using a Rayleigh-Ritz minimization of the laminate's total potential energy. A stability analysis explains the dual cylindrical shapes.

98. Hyer, M. W.: The Room-Temperature Shapes of Four-Layer Unsymmetric Cross-Ply Laminates. *Journal of Composite Materials,* vol. 16, July 1982, pp. 318–340.

82A45482

The inplane residual strains of unsymmetric laminates which have cooled from curing into a cylindrical room-temperature shape are examined numerically. Results show that the residual strains are compressive and practically independent of spatial location on the laminate. In addition, the room-temperature shapes of the four-layer unsymmetric cross-ply laminates are predicted, and it is shown that the temperature shapes are a strong function of their size and their stacking arrangement. It is demonstrated that, depending on the parameters selected, the room-temperature shape of a four-layer cross-ply unsymmetric laminate can be a unique saddle shape, a unique cylindrical shape, or a cylindrical shape that can be snapped through to another cylindrical shape.

99. Jegley, D. C.: *Effects of Thickness and Ply Orientation on Buckling of Laminated Plates.* NASA TM-87691, Feb. 1986.

86N21617#

The buckling loads of laminated plates are predicted using a new theory which takes into account transverse shearing effects. This new theory assumes trigonometric terms through-the-thickness in the displacements to take into account transverse shearing effects in thick plates. Buckling loads predicted by the new theory and by traditional theories are compared for isotropic and laminated plates. The effect of ply orientation on the buckling loads predicted by each theory is demonstrated.

100. Kibler, J. J.: *NOLIN: A Nonlinear Laminate Analysis Program.* NASA CR-2410, Feb. 1975.

75N18619#

A nonlinear, plane-stress, laminate analysis program, NOLIN, was developed which accounts for laminate nonlinearity under inplane shear and transverse extensional stress. The program determines the nonlinear stress-strain behavior of symmetric laminates subjected to any combination of inplane shear and biaxial extensional loadings. The program has the ability to treat different stress-strain behavior in tension and compression, and predicts laminate failure using any or all of maximum stress, maximum strain, and quadratic interaction failure criteria. A brief description of the program is presented including discussion of the flow of information and details of the input required. Sample problems and a complete listing of the program are also provided.

101. Knight, N. F., Jr.; and Stroud, W. J.: *Computational Structural Mechanics: A New Activity at the NASA Langley Research Center.* NASA TM-87612, Sept. 1985.

86N11540#

Complex structures considered for the late 1980's and early 1990's include composite primary aircraft structures and the space station. These structures are much more difficult to analyze than today's structures and necessitate a major upgrade in computerized structural analysis technology. A major research activity in computational structural mechanics (CSM) was initiated. The objective of the CSM activity is to develop advanced structural analysis technology that will exploit modern and emerging computers such as computers with vector and/or parallel processing capabilities. The three main research activities underway in CSM include (1) structural analysis methods development; (2) a software testbed for evaluating the methods; and (3) numerical techniques for parallel processing computers. The motivation and objectives of the CSM activity are presented and CSM activity is described. The current

CSM research thrusts and near and long term CSM research thrusts are outlined.

102. Nemeth, M. P.: Importance of Anisotropic Bending Stiffnesses on Buckling of Symmetrically Laminated Composite Plates Loaded in Compression. *26th AIAA Structures, Structural Dynamics and Materials Conference* (Orlando, Fla., Apr. 15–17), *Technical Papers, Part 1,* 1985, pp. 267–273. (Available as AIAA Paper 85-0673.)

85A30257#

The differential equation governing buckling of symmetrically laminated composite plates loaded in compression is presented in nondimensional form. From this equation, nondimensional material coefficients are obtained and a nondimensional parameter is presented that is used to assess when anisotropic bending stiffnesses can be neglected in a buckling analysis. Results obtained using finite element analyses are presented that show how boundary conditions, aspect ratio, fiber orientation, stacking sequence, and thickness affect the importance of the anisotropic bending stiffnesses.

103. Noor, A. K.; Mathers, M. D.; and Anderson, M. S.: Exploiting Symmetries for Efficient Post-Buckling Analysis of Composite Plates. *Proceedings of the 17th AIAA Structures, Structural Dynamics and Materials Conference* (King of Prussia, Pa., May 5–7), 1976, pp. 39–51.

76A30007#

Two aspects of the post-buckling analysis of composite plates are considered in this paper. The first pertains to identifying the different types of symmetry exhibited by the pre- and post-buckling responses of rectangular and skewed composite plates. A procedure is presented for exploiting these symmetries in finite-element and finite-difference analyses. The procedure can be used with existing programs having multipoint constraint capability. The second aspect pertains to the post-buckling response of biaxially loaded composite plates. Numerical results from finite-element analysis are presented which show the range of parameters for which shear deformation and/or anisotropy are important in buckling and post-buckling analysis.

104. Noor, A. K.; and Andersen, C. M.: Computerized Symbolic Manipulation in Structural Mechanics. *Trends in Computerized Structural Analysis and Synthesis* (Proceedings of Symposium, Washington, D.C., Oct. 30–Nov. 1), Pergamon Press, 1978, pp. 95–118.

79A22934

Status and recent applications of computerized symbolic manipulation to structural mechanics problems are summarized. The applications discussed include (1) generation of characteristic arrays of finite elements; (2) evaluation of effective stiffness and mass coefficients of continuum models for repetitive lattice structures; and (3) application of Rayleigh-Ritz technique to free vibration analysis of laminated composite elliptic plates. The major advantages of using computerized symbolic manipulation in each of these applications are outlined.

105. Noor, A. K.; and Mathers, M. D.: *Shear-Flexible Finite-Element Models of Laminated Composite Plates and Shells.* NASA TN D-8044, Dec. 1975.

76N14522#

Several finite-element models are applied to the linear static, stability, and vibration analysis of laminated composite plates and shells. The study is based on linear shallow-shell theory, with the effects of shear deformation, anisotropic material behavior, and bending-extensional coupling included. Both stiffness (displacement) and mixed finite-element models are considered. Discussion is focused on the effects of shear deformation and anisotropic material behavior on the accuracy and convergence of different finite-element models. Numerical studies are presented which show the effects of increasing the order of the approximating polynomials, adding internal degrees of freedom, and using derivatives of generalized displacements as nodal parameters.

106. Phan, N. D.; and Reddy, J. N.: Analysis of Laminated Composite Plates Using a Higher-Order Shear Deformation Theory. *International Journal for Numerical Methods in Engineering,* vol. 21, Dec. 1985, pp. 2201–2219.

86A21546

A higher-order deformation theory is used to analyse laminated anisotropic composite plates for deflections, stresses, natural frequencies, and buckling loads. The theory accounts for parabolic distribution of the transverse shear stresses and requires no shear correction coefficients. A displacement finite element model of the theory is developed, and applications of the element to bending, vibration, and stability of laminated plates are discussed. The present solutions are compared with those obtained using the classical plate theory and the three-dimensional elasticity theory.

107. Pindera, M. J.; and Herakovich, C. T.: An Endochronic Model for the Response of Unidirectional Composites Under Off-Axis Tensile Load. *Mechanics of Composite Materials: Recent Advances* (Proceedings of Symposium, Blacksburg, Va., Aug. 16–19, 1982), Pergamon Press, 1983, pp. 367–381.

84A33394

A methodology of modelling both nonlinear elastic and dissipative response of transversely isotropic fibrous composites is developed and illustrated with the aid of the observed response of graphite-polyimide off-axis coupons. The methodology is based on the internal variable formalism employed within the text of classical irreversible thermodynamics and entails extension of Valanis' endochronic theory to transversely isotropic media. Applicability of the theory to prediction of various response characteristics of fibrous composites is illustrated by accurately modelling such phenomena as: stiffening reversible behavior along fiber direction; dissipative response in shear and transverse tension characterized by power laws with different hardening exponents; permanent strain accumulation; nonlinear unloading and reloading; and stress-interaction effects.

108. Pindera, M. J.; and Herakovich, C. T.: An Elastic Potential for the Nonlinear Response of Unidirectional Graphite Composites. *ASME Transactions, Journal of Applied Mechanics,* vol. 51, Sept. 1984, pp. 546–550. (Available as ASME Paper 84-WA/APM-5.)

84A48986#

An elastic potential is proposed that is capable of modeling the reversible portion of the observed nonlinear response of unidirectional graphite fiber composites. The model includes both the stiffening stress-strain behavior as well as the softening Poisson's response for loading in the fiber direction. The model is compared with experimental results for Celion 6000/PMR-15 graphite-polyimide.

109. Post, D.; Czarnek, R.; Joh, D.; and Wood, J.: *Deformation Measurements of Composite Multi-Span Beam Shear Specimens by Moiré Interferometry.* NASA CR-3844, Nov. 1984.

85N11135

Experimental analyses were performed for determination of in-plane deformations and shear strains in unidirectional and quasi-isotropic graphite-epoxy beams. Forty-eight ply beams were subject to 5-point and 3-point flexures. Whole field measurements were recorded at load levels from about 20 percent to more than 90 percent of failure loads. Contour maps of U and W displacement fields were obtained by Moiré interferometry, using reference gratings of 2400 lines/mm. Clearly defined fringes with fringe orders exceeding 1000 were obtained. Whole field contour maps of shear strains were obtained by a method developed for these tests.

110. Putcha, N. S.; and Reddy, J. N.: On Dynamics of Laminated Anisotropic Plates Using a Refined Mixed Plate Element. *1984 Advances in Aerospace Sciences and Engineering: Structures, Materials, Dynamics, and Space Station Propulsion* (Proceedings of Winter Annual Meeting, New Orleans, La., Dec. 9–14), ASME, 1984, pp. 161–169.

86A11345#

An analysis of the free vibration and transient response of laminates using a mixed shear flexible element is presented. The element is based on a refined theory that accounts for a parabolic distribution of the transverse shear stresses through each lamina and requires no shear correction coefficients. The mixed formulation of the theory treats the five displacement functions and six stress resultants independently so that they are nodal degress of freedom in the finite element model. Sample problems using the element are analyzed and the results of free vibration and transient response are presented. Compared to the existing shear deformation plate theory, the present refined theory gives more accurate results for frequencies, displacements, and stresses.

111. Putcha, N. S.; and Reddy, J. N.: A Refined Mixed Shear Flexible Finite Element for the Nonlinear Analysis of Laminated Plates. *Computers and Structures,* vol. 22, no. 4, 1986, pp. 529–538.

86A28727

The present study is concerned with the development of a mixed shear flexible finite element with relaxed continuity for the geometrically linear and nonlinear analysis of laminated anisotropic plates. The formulation of the element is based on a refined higher-order theory. This theory satisfies the zero transverse shear stress boundary conditions on the top and bottom faces of the plate. Shear correction coefficients are not needed. The developed element consists of 11 degrees-of-freedom per node, taking into account three displacements, two rotations, and six moment resultants. An evaluation of the element is conducted with respect to the accuracy obtained in the bending of laminated anisotropic rectangular plates with different lamination schemes, loadings, and boundary conditions.

112. Ramnath, V.: Elastic Properties of Woven Fabric Reinforced Composites. *26th AIAA Structures, Structural Dynamics and Materials Conference* (Orlando, Fla., Apr. 15–17), *Technical Papers, Part 1*, 1985, pp. 420–425. (Available as AIAA Paper 85-0720.)

85A30274#

An analytical model for the realistic representation of a woven fabric reinforced composite is presented in this paper. The approach uses a variable cross-section geometric model in order to achieve geometric compatibility at the yarn cross-over regions. Admissible displacement and stress fields are used to determine bounds on the fabric elastic properties. The approach adopted enables the determination of the complete three-dimensional woven fabric composite properties. The in-plane fabric properties obtained through this aproach have been compared with results obtained from other approaches existing in the literature. Also, comparisons made with available experimental data indicate good agreement.

113. Reddy, J. N.; and Liu, C. F.: A Higher-Order Shear Deformation Theory of Laminated Elastic Shells. *International Journal of Engineering Science,* vol. 23, no. 3, 1985, pp. 319–330.

85A35213

A higher-order shear deformation theory of elastic shells is developed for shells laminated of orthotropic layers. The theory is a modification of the Sanders' theory and accounts for parabolic distribution of the transverse shear strains through the thickness of the shell and tangential stress-free boundary conditions on the boundary surfaces of the shell. The Navier-type exact solutions for bending and natural vibration are presented for cylindrical and spherical shells under simply supported boundary conditions.

114. Reddy, J. N.; and Phan, N. D.: Stability and Vibration of Isotropic, Orthotropic and Laminated Plates According to a Higher-Order Shear Deformation Theory. *Journal of Sound and Vibration,* vol. 98, Jan. 22, 1985, pp. 157–170.

85A34755

A higher-order shear deformation theory is used to demonstrate the natural frequencies and buckling loads of elastic plates. The theory accounts for parabolic distribution of the transverse shear strains through the thickness of the plate and rotary inertia. Exact solutions of simply supported plates are obtained and the results are compared with the exact solutions of three-dimensional elasticity theory, the first-order shear deformation theory, and the classical plate theory. The present theory predicts the frequencies and buckling loads more accurately when compared to the first-order and classical plate theories.

115. Reddy, J. N.: A Simple Higher-Order Theory for Laminated Composite Plates. *ASME Transactions, Journal of Applied Mechanics,* vol. 51, Dec. 1984, pp. 745–752. (Available as ASME Paper 84-WA/APM-34.)

85A17029#

A higher-order shear deformation theory of laminated composite plates is developed. The theory contains the same dependent unknowns as in the first-order shear deformation theory of Whitney and Pagano (1970), but accounts for parabolic distribution of the transverse shear strains through the thickness of the plate. Exact closed-form solutions of symmetric cross-ply laminates are obtained and the results are compared with three-dimensional elasticity solutions and first-order shear deformation theory solutions. The present theory predicts the deflections and stresses more accurately when compared to the first-order theory.

116. Reddy, J. N.: *A Refined Shear Deformation Theory for the Analysis of Laminated Plates.* NASA CR-3955, Jan. 1986.

86N16608#

A refined, third-order plate theory that accounts for the transverse shear strains is presented. The Navier solutions are derived for certain simply supported cross-ply and antisymmetric angle-ply laminates, and finite-element models are developed for general laminates. The new theory does not require the shear correction factors of the first-order theory (i.e., the Reissner-Mindlin plate theory) because the transverse shear stresses are represented parabolically in the present theory. Numerical results are presented to show the accuracy of the present theory in predicting the transverse stresses. Numerical results are also presented for the nonlinear bending of plates, and the results compare well with the experimental results available in the literature.

117. Robinson, J. C.; and Blackburn, C. L.: *Evaluation of a Hydrid, Anisotropic, Multilayered, Quadrilateral Finite Element.* NASA TP-1236, Aug. 1978.

78N28482#

A multilayered finite element with bending-extensional coupling is evaluated for (1) buckling of general laminated plates; (2) thermal stresses of laminated plates cured at elevated temperatures; (3) displacements of a bimetallic beam; and (4) displacement and stresses of a single-cell box beam with warped cover panels. Also, displacements and stresses for flat and spherical orthotropic and anisotropic segments are compared with results from higher order plate and shell finite-element analyses.

118. Sawyer, J. W.: *Flutter of Laminated Plates in Supersonic Flow.* NASA TM X-72800, Nov. 1975.

76N13016#

A solution procedure was developed using linear small deflection theory for the flutter of simply supported laminated plates. For such plates, the bending and extensional governing equations are coupled and have cross-stiffness terms which do not appear in classical plate theory. An extended Galerkin method is used to obtain approximate solutions to the governing equations, and the aerodynamic pressure loading used in the analysis is that given by linear piston theory with flow at arbitrary cross-flow angle. A limited parametric study was conducted for typical laminated composite plates. The calculations show that both the bending-extensional coupling and the cross-stiffness terms have a large destabilizing effect on flutter. Since classical plate theory does not consider bending-extensional coupling and cross-stiffness terms, it usually gives inaccurate and nonconservative flutter boundaries for laminated plates.

119. Sawyer, J. W.: Flutter and Buckling of General Laminated Plates. *Proceedings of the 17th AIAA Structures, Structural Dynamics and Materials Conference* (King of Prussia, Pa., May 5–7), 1976, pp. 105–112.

76A30014#

An anisotropic buckling and flutter analysis is developed with allowance for both bending-extensional coupling and bending-twisting coupling within the framework of linear small deflection theory for simply supported general laminated plates. The extended Galerkin method is used to obtain approximate solutions to the coupled governing equations. The effects of various anisotropic stiffness parameters on the static and dynamic stability of laminated plates are evaluated, with particular emphasis on assessing the range of applicability of classical orthotropic plate theory. It is shown that bending-extensional coupling and bending-twisting stiffness terms have a destabilizing effect on buckling and flutter, the effect

being more pronounced for a small number of layers. For symmetric plates, the number of layers required for orthotropic plate theory to be applicable is generally less for the buckling problem than for flutter. For square plates, aligning the fibers with the direction of airflow over the plate surface results in the highest flutter dynamic pressure.

120. Schmit, L. A.; and Fleury, D.: An Improved Analysis/Synthesis Capability Based on Dual Methods—ACCESS 3. *20th AIAA Structures, Structural Dynamics and Materials Conference* (St. Louis, Mo., Apr. 4–6), *Technical Papers on Dynamics and Loads,* 1979, pp. 23–50. (Available as AIAA Paper 79-0721.)

79A28254#

Approximation concepts and dual method algorithms are combined to create a new method for minimum weight design of structural systems. Approximation concepts convert the basic mathematical programming statement of the structural synthesis problem into a sequence of explicit primal problems of separable form. These problems are solved by constructing explicit dual functions, which are maximized subject to nonnegativity constraints. The dual method is successfully extended to deal with pure discrete and mixed continuous-discrete design variable problems. The power of the method presented is illustrated with numerical results for example problems, including a thin delta wing with fiber composite skins.

121. Schmit, L. A.; and Mehrinfar, M.: Multilevel Optimum Design of Structures With Fiber-Composite Stiffened-Panel Components. *AIAA Journal,* vol. 20, Jan. 1982, pp. 138–147. (Available as AIAA Paper 80-0723.)

82A17594#

The multilevel approach to minimum weight structural design is extended to wing box structures with fiber-composite stiffened-panel components. Strength, deflection, and panel buckling constraints are treated at the system level with equivalent-thickness-type design variables. Local buckling and panel buckling constraints are guarded against at the component level, employing detailed component dimensions as design variables. A key feature of the method is selection of change in stiffness as the component level objective function to be minimized. Numerical results are given for wing box structures with sandwich and hat-stiffened fiber-composite panels.

122. Seide, P.; and Chang, P. N. H.: *Finite Element Analysis of Laminated Plates and Shells, Volume 1.* NASA CR-157106, Mar. 1978.

78N24577

The finite element method is used to investigate the static behavior of laminated composite flat plates and cylindrical shells. The analysis incorporates the effects of transverse shear deformation in each layer through the assumption that the normals to the undeformed layer midsurface remain straight but need not be normal to the midsurface after deformation. A digital computer program was developed to perform the required computations. The program includes a very efficient equation solution code which permits the analysis of large size problems. The method is applied to the problem of stretching and bending of a perforated curved plate.

123. Shuart, M. J.: *Short-Wavelength Buckling and Shear Failures for Compression-Loaded Composite Laminates.* NASA TM-87640, Nov. 1985.

86N16271

The short-wavelength buckling (or the microbuckling) and the interlaminar and inplane shear failures of multi-directional composite laminates loaded in uniaxial compression are investigated. A laminate model is presented that idealizes each lamina. The fibers in the lamina are modeled as a plate, and the matrix in the lamina is modeled as an elastic foundation. The out-of-plane w displacement for each plate is expressed as a trigonometric series in the half-wavelength of the mode shape for laminate short-wavelength buckling. Nonlinear strain-displacement relations are used. The model is applied to symmetric laminates having linear material behavior. The laminates are loaded in uniform end shortening and are simply supported. A linear analysis is used to determine the laminate stress, strain, and mode shape when short-wavelength buckling occurs. The equations for the laminate compressive stress at short-wavelength buckling are dominated by matrix contributions.

124. Sobieszczanski-Sobieski, J.: *An Integrated Computer Procedure for Sizing Composite Airframe Structures.* NASA TP-1300, Feb. 1979.

79N17580#

A computerized algorithm to generate cross-sectional dimensions and fiber orientations for composite airframe structures is described, and its application in a wing structural synthesis is established. The algorithm unifies computations of aeroelastic loads, stresses, and deflections, as well as optimal structural sizing and fiber orientations in an open-ended system of integrated computer programs. A finite-element analysis and a mathematical-optimization technique are discussed.

125. Srinivas, S.: Wave Propagation in Laminated Orthotropic Circular Cylindrical Shells. *Recent Advances in Engineering Science, Volume 7,* Scientific Publishers, Inc., 1976, pp. 203–210.

77A39888

An exact three-dimensional analysis of wave propagation in laminated orthotropic circular cylindrical shells is developed. Numerical results are presented for three-ply shells, and for various axial wave lengths, circumferential wave numbers, and thicknesses. Results from a thin shell theory and a refined approximate theory are compared with the exact results.

126. Starnes, J. H., Jr.; and Haftka, R. T.: Preliminary Design of Composite Wings for Buckling, Strength and Displacement Constraints. *19th AIAA Structures, Structural Dynamics and Materials Conference* (Bethesda, Md., Apr. 3–5), *Technical Papers,* 1978, pp. 1–13. (Available as AIAA Paper 78-466.)

An unstiffened panel buckling constraint for balanced, symmetric laminated composites is included on the global design level in a mathematical programming structural optimization procedure for designing wing structures. Constraints are introduced by penalty functions, and Newton's method based on approximate second derivatives of the penalty terms is used as the search algorithm to obtain minimum-mass designs. Constraint approximations used during the optimization process contribute to the computational efficiency of the procedure. A criterion is developed that identifies the appropriate conservative form of the constraint approximations that are used with the optimization procedure. Minimum-mass design results are obtained for a multispar high-aspect-ratio wing subjected to material strength, minimum-gage, displacement, panel buckling and twist constraints.

127. Starnes, J. H., Jr.; and Haftka, R. T.: Preliminary Design of Composite Wing-Box Structures for Global Damage Tolerance. *21st AIAA Structures, Structural Dynamics and Materials Conference* (Seattle, Wash., May 12–14), *Technical Papers, Part 2,* 1980, pp. 529–538. (Available as AIAA Paper 80-0755.)

A procedure is presented that incorporates the influence of potential global damage conditions into the design process for minimum-mass wing-box structures. The procedure is based on mathematical-programming optimization techniques. Material-strength, minimum-gage, and panel-buckling constraints are introduced by penalty functions, and Newton's method with approximate second derivatives of the penalty terms is used as the search algorithm to obtain minimum-mass designs. A potential global damage condition is represented by a structural model with the damaged components removed. Example minimum-mass designs are obtained that simultaneously satisfy the constraints of the damaged and undamaged configurations of both graphite-epoxy and aluminum wing-box structural models.

128. Stein, M.: Nonlinear Theory for Laminated and Thick Plates and Shells Including the Effects of Transverse Shearing. *26th AIAA Structures, Structural Dynamics and Materials Conference* (Orlando, Fla., Apr. 15–17), *Technical Papers, Part 1*, 1985, pp. 259–266. (Available as AIAA Paper 85-0671.)

85A30256#

Nonlinear strain displacement relations for three-dimensional elasticity are determined in orthogonal curvilinear coordinates. To develop a two-dimensional theory, the displacements are expressed by trigonometric series representation through-the-thickness. The nonlinear strain-displacement relations are expanded into series which contain all first and second degree terms. In the series for the displacements only the first few terms are retained. Insertion of the expansions into the three-dimensional virtual work expression leads to nonlinear equations of equilibrium for laminated and thick plates and shells that include the effects of tranverse shearing. Equations of equilibrium and buckling equations are derived for flat plates and cylindrical shells. The shell equations reduce to conventional transverse shearing shell equations when the effects of the trigonometric terms are omitted and to classical shell equations when the trigonometric terms are omitted and the shell is assumed to be thin.

129. Stein, M.; and Jegley, D. C.: Effects of Transverse Shearing on Cylindrical Bending, Vibration, and Buckling of Laminated Plates. *26th AIAA Structures, Structural Dynamics and Materials Conference* (Orlando, Fla., Apr. 15–17), *Technical Papers, Part 1*, 1985, pp. 505–515. (Available as AIAA Paper 85-0744.)

85A30284#

The displacements for cylindrical bending and stretching of laminated and thick plates are expressed through-the-thickness by a few algebraic terms and a complete set of trigonometric terms. Only a few terms of this series are needed to get sufficiently accurate results for laminated and thick plates. Equations of equilibrium based on a sufficient number of terms of this series for displacements are determined using variational theorems from three-dimensional elasticity. Several examples are worked out. The displacements and stresses are obtained for simply supported isotropic and layered beams with a rectangular cross section and a sinusoidal lateral load distribution. These results are compared to an exact elasticity solution.

130. Stein, M.: Postbuckling of Long Orthotropic Plates Loaded in Combined Transverse Compression and Longitudinal Compression or Shear. *25th AIAA Structures, Structural Dynamics and Materials Conference* (Palm Springs, Calif., May 14–16), *Technical Papers, Part 1*, 1984, pp. 444–451. (Available as AIAA Paper 84-0890.)

84A31673#

The postbuckling behavior of simply supported, rectangular, isotropic and orthotropic composite plates under combined loading is determined analytically. Results are presented for a long plate loaded beyond its buckling load in either a combination of longitudinal and transverse compression or a combination of transverse compression and inplane shear. Results show that orthotropic plates may behave differently than isotropic plates for these loading conditions. Results also show that the postbuckling behavior of plates in shear with constrained edge displacements is markedly different than their behavior with stress free edges.

131. Stein, M.; and Williams, J. G.: *Buckling and Structural Efficiency of Sandwich-Blade Stiffened Composite Compression Panels.* NASA TP-1269, Sept. 1978.

78N32492#

The minimum mass structural efficiency curve was determined for sandwich-blade stiffened composite compression panels subjected to buckling and strength constraints. High structural efficiencies are attainable for this type of construction. A method of analysis is presented for the buckling of panels of this configuration which shows that buckling of such panels is strongly dependent on the through-the-thickness transverse shearing of the stiffener. Experimental results are presented and compared with theory.

132. Stein, M.: Postbuckling of Long Orthotropic Plates in Combined Shear and Compression. *24th AIAA Structures, Structural Dynamics and Materials Conference* (Lake Tahoe, Nev., May 2–4), *Collection of Technical Papers, Part 1*, 1983, pp. 310–318. (Available as AIAA Paper 83-0876.)

83A29760#

The nonlinear large-deflection partial differential equations of von Karman for orthotropic plates loaded in combined shear and compression are converted into a set of first-order nonlinear ordinary differential equations by assuming trigonometric functions in one direction. These equations are solved numerically using a two point boundary problem solver which makes use of Newton's method. Results are obtained which determine the postbuckling behavior of rectangular plates with loading up to about three times the buckling load. Both isotropic and orthotropic composite plates are considered. Results show that orthotropic plates may behave quite differently than isotropic plates and that in-plane boundary conditions are important for plates loaded in shear.

133. Stein, M.: Postbuckling of Orthotropic Composite Plates Loaded in Compression. *23rd AIAA Structures, Structural Dynamics and Materials Conference* (New Orleans, La., May 10–12), *Collection of Technical Papers, Part 1*, 1982, pp. 479–486. (Available as AIAA Paper 82-0778.)

82A30124#

The nonlinear large deflection equations of von Karman are written for "specially" orthotropic plates. The equations are then manipulated to determine the parameters required to establish postbuckling behavior. It is found that only two new parameters are needed beyond those required for buckling. By assuming trigonometric functions in one direction, the plate equations are converted into ordinary nonlinear differential equations which are solved numerically using a two point boundary problem solver that makes use of Newton's method. The postbuckling behavior is obtained for simply supported and clamped, long, rectangular, orthotropic plates covering the complete range of dimensions and material properties.

134. Stein, M.; and Stein, P. A.: *A Solution Procedure for Behavior of Thick Plates on a Nonlinear Foundation and Postbuckling Behavior of Long Plates.* NASA TP-2174, Sept. 1978.

83N34373#

Approximate solutions for three nonlinear orthotropic plate problems are presented: (1) a thick plate attached to a pad having nonlinear material properties which, in turn, is attached to a substructure which is then deformed; (2) a long plate loaded in inplane longitudinal compression beyond its buckling load; and (3) a long plate loaded in inplane shear beyond its buckling load. For all three problems, the two dimensional plate equations are reduced to one dimensional equations in the y-direction by using a one dimensional trigonometric approximation in the x-direction. Each problem uses different trigonometric terms. Solutions are obtained using an existing algorithm for simultaneous, first order, nonlinear, ordinary differential equations subject to two point boundary conditions. Ordinary differential equations are derived to determine the variable coefficients of the trigonometric terms.

135. Stroud, W. J.; Greene, W. H.; and Anderson, M. S.: Current Research on Shear Buckling and Thermal Loads With PASCO—Panel Analysis and Sizing Code. *New Directions in Optimum Structural Design,* Wiley-Interscience, 1984, pp. 663–682.

85A48712

The stiffened composite structural panel analysis and sizing code designated "PASCO" encompasses both the generality required for the exploitation of composite materials' design flexibility and an accurate buckling analysis for the detection of complex buckling modes. PASCO can accordingly design for buckling, frequency, material strength, and panel stiffness requirements. Attention is given to an additional thermal loading design capability. Design studies illustrate the importance of the multiple load condition capability when thermal loads are present.

136. Stroud, W. J.; Greene, W. H.; and Anderson, M. S.: *Buckling Loads of Stiffened Panels Subjected to Combined Longitudinal Compression and Shear: Results Obtained With PASCO, EAL, and STAGS Computer Programs.* NASA TP-2215, Jan. 1984.

84N16585#

Buckling analyses used in PASCO are summarized with emphasis placed on the shear buckling analyses. The PASCO buckling analyses included the basic VIPASA analysis, which is essentially exact for longitudinal and transverse loads, and a smeared stiffener solution, which treats a stiffened panel as an orthotropic plate. Buckling results are then presented for seven stiffened panels loaded by combinations of longitudinal compression and shear. The buckling results were obtained with the PASCO,

EAL, and STAGS computer programs. The EAL and STAGS solutions were obtained with a fine finite element mesh and are very accurate. These finite element solutions together with the PASCO results for pure longitudinal compression provide benchmark calculations to evalutate other analysis procedures.

137. Stroud, W. J.: *Optimization of Composite Structures.* NASA TM-84544, Aug. 1982.

83N10447#

Structural optimization is introduced and examples which illustrate potential problems associated with optimized structures are presented. Optimized structures may have very low load-carrying ability for an off design condition. They tend to have multiple modes of failure occurring simultaneously and can, therefore, be sensitive to imperfections. Because composite materials provide more design variables than do metals, they allow for more refined tailoring and more extensive optimization. As a result, optimized composite structures can be especially susceptible to these problems.

138. Stroud, W. J.; and Anderson, M. S.: *PASCO: Structural Panel Analysis and Sizing Code, Capability and Analytical Foundations.* NASA TM-80181, Oct. 1980.

82N21603#

A computer code denoted PASCO which can be used for analyzing and sizing uniaxially-stiffened composite panels is described. Buckling and vibration analyses are carried out with a linked-plate analysis computer code denoted VIPASA, which is incorporated in PASCO. Sizing is based on nonlinear mathematical programming techniques and employs a computer code denoted CONMIN, also incorporated in PASCO. Design requirements considered are initial buckling, material strength, stiffness, and vibration frequency. The capability of the PASCO computer code and the approach used in the structural analysis and sizing are described.

139. Stroud, W. J.; Sobieszczanski-Sobieski, J.; Walz, J. E.; and Bush, H. G.: Computerized Structural Sizing at NASA Langley Research Center. AIAA Paper 78-1550, 1978.

78A45613#

Programs at the NASA Langley Research Center associated with the development of computerized structural sizing technology are reviewed. Particular attention is given to (1) lightweight columns for space structure applications, (2) stiffened composite panels for aerospace structures, (3) thermal structures

for high-speed aircraft and space vehicles, (4) structural sizing methodology for finite-element structural models, and (5) the sizing of large complex structural systems in multidisciplinary environments. Improvements to computational efficiency are noted with reference to a reduced number of sizing variables, a reduced number of constraints, and improved sizing algorithms.

140. Stroud, W. J.; Anderson, M. S.; and Hennessy, K. W.: *Effect of Bow-Type Initial Imperfection on the Buckling Load and Mass of Graphite-Epoxy Blade-Stiffened Panels.* NASA TM-74063, Aug. 1977.

77N31542

A structural synthesis computer code which accounts for first order effects of an initial bow and which can be used for sizing stiffened composite panels having an arbitrary cross section is used to study graphite blade-stiffened panels. The effect of a small initial bow on both the load carrying ability of panels and on the mass of panels designed to carry a specified load is examined. Large reductions in the buckling load caused by a small initial bow emphasize the need for considering a bow when a panel is designed.

141. Stroud, W. J.; Agranoff, N.; and Anderson, M. S.: *Minimum-Mass Design of Filamentary Composite Panels Under Combined Loads: Design Procedure Based on a Rigorous Buckling Analysis.* NASA TN D-8417, July 1977.

77N28524#

A procedure is presented for designing uniaxially stiffened panels made of composite material and subjected to combined inplane load. The procedure used a rigorous buckling analysis and nonlinear mathematical programing techniques. Design studies carried out with the procedure consider hat-stiffened and corrugated panels made of graphite-epoxy material. Combined longitudinal compression and shear and combined longitudinal and transverse compression are the loadings used in the studies. The capability to tailor the buckling response of a panel is also explored. Finally, the adequacy of another simpler analysis-design procedure is examined.

142. Stroud, W. J.; and Agranoff, N.: *Minimum-Mass Design of Filamentary Composite Panels Under Combined Loads: Design Procedure Based on Simplified Buckling Equations.* NASA TN D-8257, Oct. 1976.

77N10552#

An analytical procedure is presented for designing hat stiffened and corrugated panels made of composite material and subjected to longitudinal (in the direction of the stiffeners) compression and shear loadings. The procedure is based on nonlinear mathematical programing techniques and a simplified set of buckling equations. Design requirements considered are buckling, strength, and extensional and shear stiffness. The effects of specified thickness, variation of cross-section dimensions, stiffness requirements, local buckling boundary conditions, and combined compression and shear loadings are shown.

143. Viswanathan, A. V.; Tamekuni, M.; and Baker, L. L.: Buckling Analysis for Anisotropic Laminated Plates Under Combined Inplane Loads. IAF Paper 74-038, 1974.

75A13687

The buckling analysis presented considers rectangular flat or curved general laminates subjected to combined inplane normal and shear loads. Linear theory is used in the analysis. All prebuckling deformations and any initial imperfections are ignored. The analysis method can be readily extended to longitudinally stiffened structures subjected to combined inplane normal and shear loads.

144. Wang, J. T. S.; and Biggers, S. B.: *Skin-Stiffener Interface Stresses in Composite Stiffened Panels.* NASA CR-172261, Jan. 1984.

84N18681#

A model and solution method for determining the normal and shear stresses in the interface between the skin and the stiffener attached flange were developed. An efficient, analytical solution procedure was developed and incorporated in a sizing code for stiffened panels. The analysis procedure described provides a means to study the effects of material and geometric design parameters on the interface stresses. These stresses include the normal stress and the shear stresses in both the longitudinal and the transverse direction.

145. Weaver, G. G., II; and Vinson, J. R.: Minimum-Mass Designs of Stiffened Graphite/Polyimide Compression Panels. *Modern Developments in Composite Materials and Structures* (Proceedings of the Winter Annual Meeting, New York, N.Y., Dec. 2-7), ASME, 1979, pp. 215-233.

80A27992#

This study presents results from an analytic investigation to determine minimum-mass designs of stiffened Gr/Pi compression panels for a wide range of uniaxial compression loads. Four panel configurations are considered: (1) hat-stiffened-laminated skin, (2) hat-stiffened-honeycomb core sandwich skin, (3) blade-stiffened-laminated skin, and (4) blade-stiffened-honeycomb core sandwich skin panels. Designs are generated by an automatic optimization computer code entitled PASCO (Panel Analysis and Sizing Code).

146. Whitcomb, J. D.: A Simple Rectangular Element for Two-Dimensional Analysis of Laminated Composites. *Computers and Structures,* vol. 22, no. 3, 1986, pp. 387-393.

86A26683

A simple rectangular finite element was developed for two-dimensional analysis of laminated composite materials. The rectangular laminated composite element eliminates the need to add elements to a model simply to account for the material properties of various laminae. This is particularly advantageous for thick laminates with many laminae. Explicit integration in terms of generalized displacements minimizes the algebraic effort required to derive the element stiffness and the thermal load vector. A substitute shape function technique is used to improve the performance of the element in modeling bending type deformation. Results for several example problems are discussed.

Structural evaluation

147. Bush, H. G.: *Experimental Evaluation of Two 36 Inch by 47 Inch Graphite/Epoxy Sandwich Shear Webs.* NASA TM X-72767, Oct. 1975.

75N33428#

The design and test are described for two large (36 in. × 47 in.) graphite/epoxy sandwich shear webs. One sandwich web was designed to exhibit strength failure of the facings at a shear load of 7638 lbs/in., which is a characteristic loading for the space shuttle orbiter main engine thrust beam structure. The second sandwich web was designed to exhibit general instability failure at a shear load of 5000 lbs/in., to identify problem areas of stability critical sandwich webs and to assess the adequacy of contemporary analysis techniques.

148. Camarda, C. J.: *Tests of Graphite/Polyimide Sandwich Panels in Uniaxial Edgewise Compression.* NASA TP-1785, Dec. 1980.

81N16129

The local and general buckling behavior of graphite/polyimide sandwich panels simply supported along all four edges and loaded in uniaxial edgewise compression was investigated. Specimens 0.635 cm thick failed by overall buckling at loads close to the analytically predicted buckling load; other panels failed by face wrinkling. Results of the wrinkling tests indicated that several buckling formulas were unconservative and therefore not suitable for design purposes; a recommended wrinkling equation is presented.

149. Davis, R. C.: *Stress Analysis and Buckling of J-Stiffened Graphite-Epoxy Panel.* NASA TP-1607, Feb. 1980.

80N18427

A graphite-epoxy shear panel with bonded on J stiffeners was investigated. Two finite element models were used to make a stress analysis of the panel. The shear load distributions in the panel from two commonly used boundary conditions, applied shear load and applied displacement, were compared with the results from one of the finite element models.

150. Davis, R. C.; and Starnes, J. H., Jr.: *Design Detail Verification Tests for a Lightly Loaded Open-Corrugation Graphite-Epoxy Cylinder.* NASA TP-1981, Mar. 1982.

82N20567#

Flat corrugated graphite-epoxy panels were tested in compression to verify selected design details of a ring-stiffened cylinder that was designed to support an axial compressive load of 157.6 kN/m without buckling. Three different sizes of subcomponent panels, with the same basic corrugation geometry, were tested. The test results indicate that the modified shell-wall design, the longitudinal joint, the load-introduction method, and the stiffener-attachment method for the proposed cylinder have adequate strength to support the design load.

151. Davis, R. C.: *Buckling Test of a 3-Meter-Diameter Corrugated Graphite-Epoxy Ring-Stiffened Cylinder.* NASA TP-2032, July 1982.

82N28665#

A 3-m-diameter by 3-m-long corrugated cylindrical shell with external stiffening rings was tested to failure by buckling. The test method was to mount the specimen as a cantilevered cylinder and apply a pure bending moment to the end. The cylinder test loading achieved was 101 percent of the design ultimate load.

152. Gurdal, Z.; Haftka, R. T.; and Starnes, J. H., Jr.: The Effect of Slots on the Buckling and Postbuckling Behavior of Laminated Plates. *Journal of Composites Technology and Research,* vol. 7, Fall 1985, pp. 82–87.

86A13870

An experimental study was conducted to evaluate the effect of lateral slots on the buckling response, the postbuckling response, and the failure characteristics of flat rectangular graphite-epoxy plates loaded in compression. The slots did not significantly affect the prebuckling and buckling behavior of the plates. The slots caused local changes in the strain distribution and out-of-plane deformations near the slot. Failure loads and modes were strongly affected by slot location.

153. Herakovich, C. T.: *Theoretical-Experimental Correlation for Buckling of Composite Cylinders Under Combined Compression and Torsion.* NASA CR-157358, July 1978.

78N29503#

Comparisons between theory and experiment for buckling of laminated graphite-epoxy and boron-epoxy cylinders under combined compression and torsion are presented. The experimental results are compared to a theory by Wu. It is shown that there is excellent agreement between theory and experiment for pure torsional loading (positive and negative), experimental buckling loads for pure compression are well below the predicted values, and good correlation is exhibited between theory and experiment for buckling under combined loading when compared in the form of normalized buckling interaction diagrams in axial-torsional load space.

154. Herakovich, C. T.; and Johnson, E. R.: Buckling of Composite Cylinders Under Combined Compression and Torsion Theoretical/Experimental Correlation. *Test Methods and Design Allowables for Fibrous Composites* (Proceedings of the Symposium, Dearborn, Mich., Oct. 2–3, 1979), American Society for Testing and Materials, 1981, pp. 341–360.

81A47816

Elastic buckling loads for laminated composite circular cylindrical shells under combined axial compression and torsion are plotted on a dimensionless load plane and determined experimentally from boron/epoxy and graphite/epoxy specimens with symmetric and asymmetric layups. Good correlation is achieved between the theoretical and experimental interaction curves. The theoretical buckling loads

are obtained from Flugge's cylindrical shell equations and for pure torsion there is good correlation, but for the pure axial compression the correlation is rather poor.

155. Howell, W. E.; and Reese, C. D.: *Reproducibility of Structural Strength and Stiffness for Graphite-Epoxy Aircraft Spoilers.* NASA TM-78723, Nov. 1978.

79N12156#

Structural strength reproducibility of graphite-epoxy composite spoilers for the Boeing 737 aircraft was evaluated by statically loading fifteen spoilers to failure at conditions simulating aerodynamic loads. Spoiler strength and stiffness data were statistically modeled using a two parameter Weibull distribution function. Shape parameter values calculated for the composite spoiler strength and stiffness were within the range of corresponding shape parameter values calculated for material property data of composite laminates. This agreement showed that reproducibility of full scale component structural properties was within the reproducibility range of data from material property tests.

156. Hyer, M. W.; and Douglas, D. O.: *Results of Graphite-Polyimide Isogrid Panel Testing.* NASA CR-153935, June 1977.

77N28229#

A procedure was developed for fabricating short-fiber HTS graphite and NR150B2 polyimide resin into an isogrid configuration. After fabrication, the panels were subjected to structural analysis and testing. The testing program is described.

157. Johnson, E. R.; Hyer, M. W.; and Carper, D. M.: Response of Long Shallow Cylindrical Panels to Radial Line Loads. *25th AIAA Structures, Structural Dynamics and Materials Conference* (Palm Springs, Calif., May 14–16), *Technical Papers, Part 1,* 1984, pp. 310–321. (Available as AIAA Paper 84-0954.)

84A31660#

The large displacement static response of shallow orthotropic panels subjected to lateral loading is examined both theoretically and experimentally. The panels are circular cylindrical open shells which are also thin and long. The straight edges are simply supported at a fixed distance apart, and the curved edges are free. The lateral load is a spatially uniform line load acting along the generator direction of the cylinder and is directed radially inward toward

the center of curvature. The load induces a circumferential thrust, and the panel can, and does, snap-through to an inverted configuration at the buckling load. The effect of load position on the response is also examined. The test panels discussed in the paper are graphite-epoxy laminates. Nominal dimensions are a radius of 60 in., a thickness of 0.060 in., and an arc length of 12 in. Very good agreement between theory and experiment was achieved.

158. Johnson, R., Jr.; Reck, R. J.; and Davis, R. C.: Design and Fabrication of a Large Graphite-Epoxy Cylindrical Shell. *19th AIAA Structures, Structural Dynamics and Materials Conference* (Bethesda, Md., Apr. 3–5), *Technical Papers,* 1978, pp. 300–310. (Available as AIAA Paper 78-507.)

78A29807#

This paper describes the design and fabrication of a 10-foot diameter by 10-foot long graphite-epoxy cylinder and reports the results of developmental tests conducted with sample joints, material coupons, and stiffening ring elements. The cylindrical shell is a ring-stiffened, open corrugation design using T300/5208 graphite-epoxy tape as the basic material for the shell wall and stiffening rings. The cylinder is designed to withstand bending loads producing the relatively low maximum load intensity of 900 lb/in. The resulting shell wall weight, including stiffening rings and fasteners, is 0.37 lb/sq ft. The shell weight expected in the graphite-epoxy cylinder represents a weight savings of approximately 23 percent over that of a comparable aluminum shell. The cylinder wall was built in three flat segments which were wrapped to the cylindrical shape upon assembly. Such an approach, made possible by the flexibility of the thin corrugated wall in a radial direction, proved to be a simple one.

159. Knight, N. F., Jr.; and Starnes, J. H., Jr.: Postbuckling Behavior of Selected Curved Stiffened Graphite-Epoxy Panels Loaded in Axial Compression. *26th AIAA Structures, Structural Dynamics and Materials Conference* (Orlando, Fla., Apr. 15–17), *Technical Papers, Part 1,* 1985, pp. 565–577. (Available as AIAA Paper 85-0768.)

85A30289#

Results of an experimental and analytical study of the postbuckling behavior of selected curved stiffened graphite-epoxy panels loaded in axial compression are presented. The postbuckling response and failure characteristics of the panels are described. Each panel had four equally-spaced I-shaped stiffeners and

16-ply quasi-isotropic skins. Panels with three different stiffener spacings were tested. Failure of all panels initiated in a skin-stiffener interface region. Analytical results from a nonlinear general shell finite element analysis computer code correlate well with typical postbuckling test results up to failure. The analytical modeling detail necessary to predict accurately the response of the panel is described. Measured initial geometric imperfections were included in the postbuckling analysis.

160. Knight, N. F., Jr.; and Starnes, J. H., Jr.: Postbuckling Behavior of Axially Compressed Graphite-Epoxy Cylindrical Panels With Circular Holes. Paper presented at the ASME Joint Pressure Vessels and Piping/Applied Mechanics Conference, San Antonio, Tex., June 17–21, 1984.

84A33132#

The results of an experimental and analytical study of the effects of circular holes on the postbuckling behavior of graphite-epoxy cylindrical panels loaded in axial compression are presented. The STAGSC-1 general shell analysis computer code is used to determine the buckling and postbuckling response of the panels. The unstable equilibrium path of the postbuckling response is obtained analytically by using a method based on controlling an equilibrium-path-arc-length parameter instead of the traditional load parameter. The effects of hole diameter, panel radius, and panel thickness on postbuckling response are considered in the study. Experimental results are compared with the analytical results and the failure characteristics of the graphite-epoxy panels are described.

161. Morita, W. H.; and Graves, S. R.: *Graphite/LaRC-160 Technology Demonstration Segment Test Results.* NASA CR-172123, June 1983.

83N30539#

A structural test program was conducted on a Celion/LaRC-160 graphite/polyimide technology demonstration segment (TDS) to verify the technology. The 137 × 152 cm (54 × 60 in.) TDS simulates a full-scale section of the orbiter composite body flap design incorporating three ribs and extending from the forward cove back to the rear spar. The TDS was successfully subjected to mechanical loads and thermal environments (−170° to 316°C) simulating 100 shuttle orbiter missions. Successful completion of the test program verified the design, analysis, and fabrication methodology for bonded Gr/PI honeycomb sandwich structure and demonstrated that Gr/PI composite technology readiness is established.

162. Plunkett, R.: Damping Mechanisms in Fiber Reinforced Laminates. *Mechanics of Composite Materials: Recent Advances* (Proceedings of the Symposium, Blacksburg, Va., Aug. 16–19, 1982), Pergamon Press, 1983, pp. 93–104.

84A33380

Low strain damping in fiber reinforced composite materials is due to material loss factors in both fibers and matrix materials. The high modulus of the fibers makes up for the low damping factor. Strains high enough to cause transverse layer cracking in laminates with organic matrix materials cause a larger permanent increase in the damping factor. The increase is not due to the transverse cracks but rather to short microcracks in the high shear strain regions. Other damping mechanisms at high strains are also discussed.

163. Rouse, M.: Postbuckling of Flat Unstiffened Graphite-Epoxy Plates Loaded in Shear. *26th AIAA Structures, Structural Dynamics and Materials Conference* (Orlando, Fla., Apr. 15–17), *Technical Papers, Part 1,* 1985, pp. 605–616. (Available as AIAA Paper 85-0771.)

85A30292#

An experimental and analytical study was conducted of the postbuckling response and failure characteristics of 8-, 16-, and 24-ply unstiffened graphite-epoxy plates under shear loading, including orthotropic, quasi-orthotropic, entirely ±45° ply, and (0/90)s-class plates. Postbuckling stiffness is noted to be influenced by the stacking sequence and by the inplane boundary condition imposed on the test section by a picture-frame test fixture. The ratio of failure load to buckling load was higher for specimens that had higher width-to-thickness ratios. It is noted that high displacement gradients may induce transverse shear stresses and that delamination was the failure mechanism for many of the 8-ply specimens. An adhesive failure between the composite specimen and metal edge reinforcements initiated failure of the 16- and 24-ply specimens.

164. Shuart, M. J.: *Hole Interaction and Load Introduction Effects for Compression-Loaded Laminates With Holes.* NASA TM-86287, Aug. 1984.

84N32435

The effects of hole interaction and load introduction for laminates with holes were investigated. A hole interaction region was identified in some specimens, and this region was ineffective for carrying load. The membrane stiffness of a specimen with

a hole interaction region is less than the membrane stiffness of a similar specimen without a hole interaction region. A load introduction effect that occurs on the sublaminate level was also observed. This effect can cause a nonuniform load diffusion into the interior of the specimen.

165. Shuart, M. J.; and Hagaman, J. A.: *Buckling and Failure Characteristics of Graphite-Polyimide Shear Panels.* NASA TP-2153, May 1983.

83N24561#

The buckling and failure characteristics of unstiffened, blade-stiffened, and hat-stiffened graphite-polyimide shear panels are described. The picture frame shear test is used to obtain shear stress-strain data at room temperature and at 316°C. The experimental results are compared with a linear buckling analysis, and the specimen failure modes are described. The effects of the 316°C test temperature on panel behavior are discussed.

166. Shyprykevich, P.: Characterization of Graphite/Epoxy Laminates for Aeroelastic Tailoring. *Composite Materials: Testing and Design* (Proceedings of the Fifth Conference, New Orleans, La., Mar. 20–22, 1978), ASTM, 1979, pp. 40–56.

80A21130

A study of interaction between wing bending and twist by graphite/epoxy anisotropic laminates used in aircraft wing skins is presented. The laminates were used as covers for subscale box beams supported as a cantilever and tested in tip shear and tip torque, measuring beam response with a reflected light technique. The results indicated that the in-plane stiffness properties of anisotropic laminates can be predicted if the layer properties of the composite materials are known; thus, the coupled bending/twist response of wing type structures made from these laminates can be determined provided the limits of the laminate linear behavior are not significantly exceeded.

167. Starnes, J. H., Jr.; and Williams, J. G.: Failure Characteristics of Graphite-Epoxy Structural Components Loaded in Compression. *Mechanics of Composite Materials: Recent Advances* (Proceedings of the Symposium, Blacksburg, Va., Aug. 16–19, 1982), Pergamon Press, 1983, pp. 283–306.

84A33390

Failure characteristics of compressively loaded graphite-epoxy components are described. Experimental results for both strength-critical laminates and structural components with postbuckling strength are presented. Effects of low-speed impact damage and circular holes on compressive strength are discussed. Delamination and shear crippling failure mechanisms that limit the performance of strength-critical laminates are described. Transverse shear and skin-stiffener separation failure mechanisms that limit the performance of components with postbuckling strength are also described. The influence of matrix properties on compressive strength improvements for impact damaged laminates is discussed. Experimental data and results from a failure analysis for strength-critical laminates with cutouts are discussed and compared with impact damage results. Typical postbuckling test results are compared with analytical predictions.

168. Starnes, J. H., Jr.; and Rouse, M.: Postbuckling and Failure Characteristics of Selected Flat Rectangular Graphite-Epoxy Plates Loaded in Compression. *22nd AIAA Structures, Structural Dynamics and Materials Conference* (Atlanta, Ga., Apr. 6–8), *Technical Papers, Part 1*, 1981, pp. 423–434. (Available as AIAA Paper 81-0543.)

81A29422#

Results of an experimental study of the postbuckling response and failure characteristics of 16- and 24-ply quasi-isotropic and 24-ply orthotropic flat rectangular graphite-epoxy plates loaded in compression are examined. Some of the specimens had circular drilled holes or were subjected to low-speed impact damage. The ratio of failure load to buckling load was found to be higher for specimens with lower initial buckling strains than for specimens with higher initial buckling strains. Some specimens supported more than five times their initial buckling load before failing.

169. Starnes, J. H., Jr.; Knight, N. F., Jr.; and Rouse, M.: Postbuckling Behavior of Selected Flat Stiffened Graphite-Epoxy Panels Loaded in Compression. *23rd AIAA Structures, Structural Dynamics and Materials Conference* (New Orleans, La., May 10–12), *Collection of Technical Papers, Part 1*, 1982, pp. 464–478. (Available as AIAA Paper 82-0777.)

82A30123#

Results of an experimental study of the postbuckling behavior of selected flat stiffened graphite-epoxy panels loaded in compression are presented. The

postbuckling response and failure characteristics of undamaged panels and panels damaged by low-speed impact are described. Each panel had four equally-spaced I-section stiffeners and 16- or 24-ply quasi-isotropic skins. Panels with three different stiffener spacings were tested. Some undamaged specimens supported as much as three times their initial buckling load before failing. Failure of all panels was initiated in a skin-stiffener interface region. Analytical results obtained from a nonlinear general shell finite element analysis computer code correlate well with typical postbuckling test results up to failure.

170. Waters, W. A., Jr.; and Williams, J. G.: *Failure Mechanism of Laminates Transversely Loaded by Bolt Push-Through.* NASA TM-87603, Sept. 1985.

86N14314#

Stiffened composite panels proposed for fuselage and wing design utilize a variety of stiffener-to-skin attachment concepts including mechanical fasteners. One potential failure mode for bolted panels occurs when the bolts pull through the stiffener attachment flange or skin. The resulting loss of support by the skin to the stiffener and by the stiffener to the skin can result in local buckling and subsequent panel collapse. The characteristic failure modes associated with bolt push-through failure are described and the results of a parametric study of the effects that different material systems, boundary conditions, and laminates have on the forces and displacements required to cause damage and bolt push-through failure are presented.

171. Williams, J. G.; and Stein, M.: Buckling Behavior and Structural Efficiency of Open-Section Stiffened Composite Compression Panels. AIAA Paper 76-1727, 1976.

76A35202#

Several experiments with J- and blade-stiffened graphite/epoxy panels were conducted to obtain insight into how well experimental data could be correlated with analysis for the buckling behavior of open-section stiffened composite compression panels. Although some nonlinear behavior was observed during the experiment, adequate correlation with analysis was obtained to justify the use of linear, thin-plate buckling analysis in a minimum-weight design synthesis program for J- and blade-configurations.

Impact and damage tolerance.

172. Adams, D. S.; and Herakovich, C. T.: Influence of Damage on the Thermal Response of Graphite-Epoxy Laminates. *Journal of Thermal Stresses,* vol. 7, no. 1, 1984, pp. 91–103.

84A48568

A constrained-displacement finite-element approach for studying the influence of transverse cracks and delaminations on the thermal response of laminated composites is presented. Typical results are given in the form of percent-retention curves for the coefficient of thermal expansion as a function of crack density. Cross-ply and quasi-isotropic T300/5208 graphite-epoxy laminates are considered. It is shown that transverse cracks can have a significant influence on the coefficient of thermal expansion, but delaminations located symmetrically about the laminate midplane have no influence on thermal expansion.

173. Avva, V. S.: Effect of Specimen Size on the Buckling Behavior of Laminated Composites Subjected to Low-Velocity Impact. *Compression Testing of Homogeneous Materials and Composites* (Proceedings of the Symposium, Williamsburg, Va., Mar. 10–11, 1982), ASTM, 1983, pp. 140–154.

84A29893

An experimental investigation was conducted to study the effect of specimen size on the buckling strains of composite laminates subjected to low-velocity projectile impact. The specimens were fabricated from a T300/5208 graphite/epoxy material in 16- and 32-ply quasi-isotropic laminates.

174. Bigelow, C. A.: *An Assessment of Buffer Strips for Improving Damage Tolerance of Composite Laminates at Elevated Temperature.* NASA TM-83257, Dec. 1981.

82N18325#

Buffer strips greatly improve the damage tolerance of graphite/epoxy laminates loaded in tension. Graphite/polyimide buffer strip panels were made and tested to determine their residual strength at ambient and elevated (177°C) temperature. Each panel was cut in the center to represent damage. Panels were radiographed and crack-opening displacements were recorded to indicate fracture, fracture arrest, and the extent of damage in the buffer strip after arrest.

175. Bigelow, C. A.: Buffer Strips in Composites at Elevated Temperature. *Journal of Composite Materials,* vol. 17, Nov. 1983, pp. 549–560.

84A25007

The composite material "buffer strip" concept is presently investigated at elevated temperatures for the case of graphite/polyimide buffer strip panels where the buffer strip material was 0-deg S-glass/polyimide. Each panel was loaded in tension until it failed, and radiographs and crack opening displacements were recorded during the tests to determine fracture onset, fracture arrest, and the extent of damage in the buffer strip after crack arrest.

176. Byers, B. A.: *Behavior of Damaged Graphite/ Epoxy Laminates Under Compression Loading.* NASA CR-159293, Aug. 1980.

82N32420#

The influence of three different resin systems on the damage tolerance of graphite/epoxy laminates was evaluated. Testing consisted of both static compression and cyclic compression evaluation of 10.2 by 15.2 by 0.5 cm (4 by 6 by 0.2 in.) laminates with circular holes, simulated delaminations, and low-velocity impact. Damage growth under steadily increasing compression and cyclic compression loading was monitored.

177. Chen, J. K.; and Sun, C. T.: Analysis of Impact Response of Buckled Composite Laminates. *Composite Structures,* vol. 3, no. 2, 1985, pp. 97–118.

85A32676

The dynamic response of buckled composite plates impacted by a hard object is studied. In order to evaluate the contact force, an experimentally established contact law which accounts for the permanent indentation is employed. The static postbuckling problem of laminated plates is first considered. The impact responses include contact force histories, dynamic deflections, and dynamic strains in the plate for various buckling conditions. In addition, free vibration of buckled laminated plates is also solved to determine the range of natural frequencies which are needed to choose a proper time increment for time integration in the impact analysis.

178. Chen, J. K.; and Sun, C. T.: Dynamic Large Deflection Response of Composite Laminates Subjected to Impact. *Composite Structures,* vol. 4, no. 1, 1985, pp. 59–73.

85A39599

Impact responses of composite laminates with and without initial stresses are investigated using the finite element method. In the time integration, the Newmark constant acceleration algorithm is used in conjunction with successive iterations within each time step. Numerical results, including the contact force histories, deflections, and strains in the plate, are presented.

179. Daugherty, R. L.; and Howard, S. A.: Effects of Debonds on the Strength of Composite Plates. *Proceedings of the Second International Conference on Composite Materials* (Toronto, Canada, Apr. 16–20), Metallurgical Society of AIME, 1978, pp. 1224–1236.

79A17050

The effect of debonds on composite structures under compressive loads is investigated in this paper. Because of the inclusion of debonds, unsymmetric laminated rectangular plates are considered. The resulting bending-extension coupling is included and is shown to be quite significant for thin laminates typical of debonded regions. The solution technique is based on the theorem of minimum potential energy; and both simply supported and clamped boundaries are considered.

180. Dwivedi, S. N.: *Feasibility Study of Deformation Modes of Deforming Composite Materials Using the Visioplasticity Method.* NASA CR-165067, Nov. 1981.

82N15125#

The impact resistance and the effect of impact on fiber reinforced composite materials was investigated. The visioplastic method for studying deformation modes and transient impact distribution is described and the basic equations used in the visioplastic method are presented. The subroutine for digitizing the input data and a computer program (STRESS) for calculating the strains and stresses of deforming projectiles are presented.

181. Farley, G. L.: *Effect of Low-Velocity or Ballistic Impact Damage on the Strength of Thin Composite and Aluminum Shear Panels.* NASA TP-2441, May 1985.

85N26924#

Impact tests were conducted on shear panels fabricated from 6061-T6 aluminum and from woven fabric prepreg of Du Pont Kevlar fiber/epoxy resin and graphite fiber/epoxy resin. Composite panels were impacted with a 1.27-cm (0.5-in) diameter aluminum sphere at low velocities of 46 m/sec (150 ft/sec) and 67 m/sec (220 ft/sec). Ballistic impact conditions consisted of a tumbled 50-caliber projectile impacting loaded composite and aluminum shear panels. The

results of these tests indicate that ballistic threshold load (the lowest load which will result in immediate failure upon penetration by the projectile) varied between 0.44 and 0.61 of the average failure load of undamaged panels.

182. Guynn, E. G.; and O'Brien, T. K.: The Influence of Lay-Up and Thickness on Composite Impact Damage and Compression Strength. *26th AIAA Structures, Structural Dynamics and Materials Conference* (Orlando, Fla., Apr. 15–17), *Technical Papers, Part 1,* 1985, pp. 187–196. (Available as AIAA Paper 85-0646.)

85A30246#

The effects of composite stacking sequence, thickness, and percentage of zero-degree plies on the size, shape, and distribution of delamination through the laminate thickness and on residual compression strength following impact were studied. Graphite/epoxy laminates were impacted with an 0.5 inch diameter aluminum sphere at a specific low or high velocity. Impact damage was measured nondestructively by ultrasonic C-scans and X-radiography and destructively by the deply technique, and compression strength tests were performed.

183. Haftka, R. T.; Starnes, J. H., Jr.; and Nair, S.: Design for Global Damage Tolerance and Associated Mass Penalties. *Journal of Aircraft,* vol. 20, Jan. 1983, pp. 83–88.

83A15321#

A structural design with global damage tolerance is defined as a design that can tolerate the destruction of one or more major structural components. The mass penalty associated with improving the global damage tolerance of optimized structures is evaluated herein for structures typically used in aircraft wing construction. It is shown that this mass penalty is strongly related to the degree of redundancy of the structure, being most severe for structures of low redundancy. For highly redundant wingbox structures made of composite materials, it is shown that significant improvement in global damage tolerance may be achieved without any mass penalty.

184. Hertzberg, P. E.; Smith, B. W.; and Miller, A. G.: *Effect of Matrix Resin on the Impact Fracture Characteristics of Graphite-Epoxy Laminates.* NASA CR-165784, Jan. 1982.

83N13177#

The effect of resin chemistry on basic impact energy absorbent mechanisms exhibited by graphite-epoxy composites was investigated. Impact fracture modes and microscopic resin deformation characteristics were examined for 26 NASA-impacted graphite-epoxy laminates with different resin chemistries. Discrete specimen fracture modes were identified through cross sectional examination after impact and subsequently compared with measured glass transition temperatures, cure cycles, and residual impact capabilities. Microscopic resin deformation mechanisms and their overall relationship to impact loading conditions, voids, and resin content were also characterized through scanning electron microscopic examination of separated fracture surfaces.

185. Humphreys, E. A.; and Goering, J.: *Development of an Analytic Procedure To Calculate Damage Accumulation in Composites During Low-Velocity Impact.* NASA CR-166086, Feb. 1983.

83N23360#

A computerized procedure was developed to model the response of a laminated composite plate subjected to low-velocity impact. The methodology incorporated transient dynamics finite element analysis coupled with composite layer and interlaminar stress predictions. Damage was predicted using a stress based failure criteria and incorporated into the solution as stiffness modifications. The force-displacement relation between the impactor and plate was modelled with a nonlinear contact spring similar to Hertzian contact. Analyses performed predicted ply damage early in the impact event when the displacement fields were characteristic of high frequency flexure response.

186. Joshi, S. P.; and Sun, C. T.: Impact Induced Fracture in a Laminated Composite. *Journal of Composite Materials,* vol. 19, Jan. 1985, pp. 51–66.

85A30849

Experimental data of the three-dimensional problem of impact of a flat strip by a spherical impactor are presented and interpreted qualitatively by comparison with a plane-strain numerical analysis of an infinitely wide plate impacted by a cylindrical impactor. The role of transverse shear stress in proximal and middle layer crack initiation is established. A detailed presentation of damage is provided with exact delamination zones. The basic conclusions drawn establish a basis for further research in understanding impact induced fracture in composites.

187. Kelkar, A.; Elber, W.; and Raju, I. S.: Large Deflection Behavior of Quasi-Isotropic Laminates Under Low-Velocity Impact Type Point Loading. *26th AIAA Structures, Structural Dynamics and Materials Conference* (Orlando, Fla., Apr. 15–17), *Technical Papers, Part 1,* 1985, pp. 432–441. (Available as AIAA Paper 85-0723.)

85A30276#

Eight-ply quasi-isotropic circular composite plates of Thornel 300 graphite in Narmco 5208 epoxy resin (T300/5208) were analyzed to obtain the large deformation behavior under low-velocity impact type point loads. A simple plate-membrane coupling model was developed. The impact type point loads were replaced by equivalent quasi-static point loads. The plate-membrane coupling model was used to obtain the large deformation shapes for the thin circular composite laminates. The analyses indicated that the large deformation shapes of the composite plates under point loads vary with the centerpoint displacements, and hence are different for different load levels. Quasi-isotropic plates were analyzed by replacing anisotropic bending stiffness components with the equivalent flexural stiffness for the isotropic plates. The plate-membrane coupling model was verified by conducting a series of tests on clamped circular quasi-isotropic laminates. Deflected shapes for the thin composite plates were experimentally obtained. These shapes agreed well with the analytically predicted shapes.

188. Knauss, W. G.; Babcock, C. D.; and Chai, H.: Observation of Damage Growth in Compressively Loaded Laminates. *Experimental Mechanics,* vol. 23, Sept. 1983, pp. 329–337.

83A46810

Graphite/epoxy laminates have a definite advantage with respect to the strength-to-weight relation over many standard engineering materials used in aerospace applications. However, this advantage is somewhat reduced by the sensitivity of these laminates to operational hazards, which include a low-velocity impact by foreign objects. Investigations conducted by Chai (1982) and Knauss et al. (1980) have been concerned with the growth of impact damage in compressively loaded laminates and the visualization of such an impact damage. The present study represents a condensation of parts of these investigations, taking into account a determination of the damage-growth mechanism via real-time recording of the impact event. The material considered, a T300/5208 graphite/epoxy laminate, is typical of the configuration proposed for future heavily loaded primary structures.

189. Lal, K. M.: Residual Strength Assessment of Low-Velocity Impact Damage of Graphite/Epoxy Laminates. *Journal of Reinforced Plastics and Composites,* vol. 2, Oct. 1983, pp. 226–238.

84A19796

This report contains the study of low-velocity transverse impact damage of graphite/epoxy T300/5208 composite laminates. The specimens, 100-mm diameter clamped plates, were impact damaged by a cantilever-type instrumented 1-inch diameter steel ball. The study was limited to impact velocity of 6 m/sec. Rectangular strips, 50 mm × 125 mm, were cut from the impact-damaged specimens so that the impact damage zone was in the center of the strips. These strips were tested in tension to obtain their residual strength. An energy dissipation model was developed to predict the residual strength from fracture mechanics concepts.

190. Lal, K. M.: Low-Velocity Transverse Impact Behavior of 8-Ply, Graphite-Epoxy Laminates. *Journal of Reinforced Plastics and Composites,* vol. 2, Oct. 1983, pp. 216–225.

84A19795

This work discusses the behavior of eight-ply, quasi-isotropic, graphite-epoxy laminates subjected to low-velocity transverse impact loading. Large deflection theory of plates was used to predict the load-deflection characteristics during the impact event. The impact model considered that the indentation, flexural, and shear stiffnesses could be represented by three equivalent springs in series. The analysis of static and dynamic impact loading test data concluded that the membrane parameter β used in the flexural stiffness relation was proportional to the square of the coefficient of restitution.

191. Lal, K. M.: Coefficient of Restitution for Low-Velocity Transverse Impact of Thin Graphite-Epoxy Laminates. *Composites Technology Review,* vol. 6, Fall 1984, pp. 112–117.

85A14160

A simple model is presented for the low-velocity impact behavior of thin plates composed of T300/5208 graphite-epoxy. The model, which is constructed on the assumption that the plate is inextensible in the fiber direction and that the material is incompressible in the z direction, is able to predict the energy absorbed by the target during an impact event, or the "coefficient of restitution." The plate essentially deforms by shear, allowing the model to neglect bending deformations. The coefficient of restitution is predicted to increase with large

interlaminar shear strength and low transverse shear modulus of the laminate. Experimentally measured values agree with model-predicted values within a reasonable error.

192. Lal, K. M.: Relationship Between Phase Difference and Coefficient of Restitution During Low-Velocity Foreign Object Transverse Damage of Composite Plates. *Composites Technology Review,* vol. 6, Fall 1984, pp. 109–111.

85A14159

This work discusses a model to correlate the coefficient of restitution of low-velocity transverse impacts of graphite-epoxy laminates with the residual deformation or central deflection at the end of the impact event. It is assumed that the energy absorbed by the target can be calibrated in terms of residual deflection, and thereby in terms of phase difference between the occurrence of impact force and central deflection to their maximas. Analysis was modeled on the basis of the experience from impact tests. Predictions are compared with the test results of impacted circular and flat plates. Experimentally measured values of coefficient of restitution and phase difference agreed well with the predicted relationship between them.

193. Lee, J. D.; Du, S.; and Liebowitz, H.: Three-Dimensional Finite Element and Dynamic Analysis of Composite Laminate Subjected to Impact. *Computers and Structures,* vol. 19, no. 5–6, 1984, pp. 807–813.

85A22001

A three-dimensional finite element and dynamic analysis has been made for a layered fiber-reinforced composite laminate subjected to a given impact loading. The central difference method is employed in this analysis. The numerical results for the transient response of the laminate are presented.

194. Lee, J. D.: Three-Dimensional Finite Element Analysis of Layered Fiber-Reinforced Composite Materials. *Computers and Structures,* vol. 12, Sept. 1980, pp. 319–339.

81A16295

A three-dimensional finite element analysis was performed for a biaxially loaded composite laminate (with a centered hole) consisting of several fiber-reinforced composite layers each with a specified fiber orientation. The detailed stress distribution around the hole was determined. Also, the locations of initial damage zones due to different failure mechanisms were indicated.

195. Lee, J. D.: Three-Dimensional Finite-Element Analysis of Damage Accumulation in Composite Laminate. *Computers and Structures,* vol. 15, no. 3, 1982, pp. 335–350.

82A31193

A three-dimensional finite-element computer program has been developed to analyze layered fiber-reinforced composite laminate. This program is capable of (1) calculating the detailed stress distribution, (2) identifying the damage zone and mode of failure, (3) analyzing the damage accumulation, and (4) determining the ultimate strength of the composite laminate.

196. Liebowitz, H.: *The Effect of Low-Velocity Impact on the Strength Characteristics of Composite Material Laminates.* NASA CR-170059, Mar. 1983.

83N20282#

The nonlinear vibration response of a double cantilevered beam subjected to pulse loading over a central sector is studied. The initial response is generated in detail to ascertain the energetics of the response. The total energy is used as a gauge of the stability and accuracy of the solution. It is shown that to obtain accurate and stable initial solutions an extremely high spatial and time resolution is required. This requirement was only evident through an examination of the energy of the system. It is proposed, therefore, to use the total energy of the system as a necessary stability and accuracy criterion for the nonlinear response of conservative systems. The results also demonstrate that even for moderate nonlinearities, the effects of membrane forces have a significant influence on the system.

197. Palmer, R. J.: *Investigation of the Effect of Resin Material on Impact Damage to Graphite/ Epoxy Composites.* NASA CR-165677, May 1981.

81N24182#

The results of an experimental program are described which establishes the feasibility and guidelines for resin development. The objective was to identify the basic epoxy neat-resin properties that improve low-velocity impact resistance and toughness of graphite/epoxy laminates and at the same time maintain useful structural laminate mechanical properties. Materials tests from twenty-three toughened epoxy resin matrix systems are included.

198. Pang, S. S.; Zhang, Z. D.; Chern, S. S.; and Hsiao, C. C.: *Energy Absorption by Polymer Crazing.* NASA CR-173275, 1983.

84N16335#

During the past thirty years, a tremendous amount of research was done on the development of crazing in polymers. The phenomenon of crazing was recognized as an unusual deformation behavior associated with a process of molecular orientation in a solid to resist failure. The craze absorbs a fairly large amount of energy during the crazing process. When a craze does occur the surrounding bulk material is usually stretched to several hundred percent of its original dimension and creates a new phase. The total energy absorbed by a craze during the crazing process in creep was calculated analytically with the help of some experimental measurements. A comparison of the energy absorption by the new phase and that by the original bulk uncrazed medium is made.

199. Paul, J. T. Jr.; and Buntin, G. A.: *Graphite Fiber Surface Treatment To Improve Impact Strength and Fracture Resistance in Subsequent Composites.* NASA CR-165901, Dec. 1982.

83N16392#

Graphite (or carbon) fiber composite impact strength improvement was attempted by modifying the fiber surface. Elastomeric particles were made into lattices and deposited ionically on surface treated graphite fiber in an attempt to prepare a surface containing discrete rubber particles. With hard, nonelastomeric polystyrene, discrete particle coverage was achieved. All the elastomeric containing lattices resulted in elastomer flow and filament agglomeration during drying.

200. Poe, C. C., Jr.; Illg, W.; and Garber, D. P.: *A Program To Determine the Effect of Low-Velocity Impacts on the Strength of the Filament-Wound Rocket Motor Case for the Space Shuttle.* NASA TM-87588, Sept. 1985.

86N11292#

Graphite/epoxy filament-wound cases (FWC) are being developed for the solid rocket motors of the space shuttle. The 12-foot-diameter FWC cases are wound with AS4W graphite fiber impregnated with an epoxy resin and are about 1.4 inches or more thick. The impact tests are conducted on a representative filament-wound laminate. The specimens are supported to simulate a fueled (stiff) and an empty (flexible) case. Impactors of various kinetic energy, mass, and shape are used. The conditions that give minimum visual evidence of damage are emphasized. The capability to characterize impact damage with various nondestructive evaluation (NDE) methods is also evaluated. After impact, the specimens are loaded uniaxially in tension to determine residual strengths.

The residual strengths of impacted specimens are compared with fracture mechanics predictions based on a semi-elliptic surface cut of the same size as the impact damage.

201. Poe, C. C., Jr.; and Kennedy, J. M.: An Assessment of Buffer Strips for Improving Damage Tolerance of Composite Laminates. *Journal of Composite Materials Supplement,* vol. 14, no. 1, 1980, pp. 57–70.

81A23753#

Graphite/epoxy panels with buffer strips were tested in tension to measure their residual strength with crack-like damage. The buffer strips arrested the cracks and increased the residual strengths significantly over those of plain laminates without buffer strips. A shear-lag type stress analysis correctly predicted the effects of layup, buffer material, buffer strip width and spacing, and the number of plies of buffer material.

202. Poe, C. C., Jr.: *Tensile Strength of Composite Sheets With Unidirectional Stringers and Crack-Like Damage.* NASA TM-86310, Sept. 1984.

85N11139#

The residual strength of composite sheets with bonded composite stringers loaded in tension was determined. The results are summarized. About 50 graphite/epoxy composite panels with crack-like slots were monotonically loaded in tension to failure. Both sheet layup and stringer configuration were varied. The composite panels have considerable damage tolerance. The stringers arrested cracks that ran from the crack-like slots, and the residual strengths were considerably greater than those of unstifffened composite sheets.

203. Rhodes, M. D.; Williams, J. G.; and Starnes, J. H., Jr.: Low-Velocity Impact Damage in Graphite-Fiber Reinforced Epoxy Laminates. *Reinforcing the Future* (Proceedings of the Thirty-Fourth Annual Conference, New Orleans, La., Jan. 30–Feb. 2), Society of the Plastics Industry, Inc., 1979, pp. 20-D1–20-D10.

79A31032#

An experimental investigation was conducted to identify the failure mechanisms and to understand damage propagation in compression-loaded composite structures. The tests were conducted on several laminates of different ply orientation with thickness that ranged from 0.56 to 0.75 cm. The panels were damaged by 1.27-cm-diameter aluminum

spheres propelled normal to the specimen surface at velocities ranging from 30 m/s to 140 m/s. Results indicate that there is significant internal laminate damage due to low-velocity impact with no surface damage. The internal damage consists of delamination and intraply cracking. Three damage propagation modes were identified as causing specimen failure, which are delamination, axial load-lateral deformation coupling, and local shear failure.

204. Rhodes, M. D.; Williams, J. G.; and Starnes, J. H., Jr.: Effect of Impact Damage on the Compression Strength of Filamentary-Composite Hat-Stiffened Panels. *Selective Application of Materials for Products and Energy* (Proceedings of the Twenty-Third National Symposium and Exhibition, Anaheim, Calif., May 2–4), Society for the Advancement of Material and Process Engineering, 1978, pp. 300–319.

79A20818

An experimental investigation was conducted to evaluate the effect of low-velocity impact damage on the compression strength of filamentary-composite hat-stiffened panels. Twenty-four specimens of three design configurations fabricated from graphite-epoxy and boron-epoxy materials were tested. All three design configurations met the design buckling requirements in the undamaged condition. The impact damage was caused by firing aluminum spheres at the panels to simulate impact on aircraft from runway stones. Test results suggest that impact damage may be more dependent on the matrix properties than on the fiber properties of the composite materials considered.

205. Rhodes, M. D.: *Impact Tests on Fibrous Composite Sandwich Structures.* NASA TM-78719, Oct. 1978.

78N33152

The effect of low-velocity impact on the strength of laminates fabricated from graphite/epoxy and Kevlar 49/epoxy composite materials was studied. The test laminates were loaded statically either in uniaxial tension or compression when impact occurred to evaluate the effect of loading on the initiation of damage and/or failure. Typical aircraft service conditions such as runway debris encountered during landing were simulated by impacting 1.27-cm-diameter projectiles normal to the plane of the test laminates at velocities between 5.2 and 48.8 m/s.

206. Rhodes, M. D.; and Williams, J. G.: Concepts for Improving the Damage Tolerance of Composite Compression Panels. Paper presented at U.S.

Department of Defense and NASA 5th Conference on Fibrous Composites in Structural Design, New Orleans, La., Jan. 27–29, 1981.

81A32825#

The results of an experimental evaluation of graphite-epoxy composite compression panel impact damage tolerance and damage propagation arrest concepts are reported. The tests were conducted on flat plate specimens and blade-stiffened structural panels such as those used in commercial aircraft wings, and the residual strength of damaged specimens and their sensitivity to damage while subjected to in-plane compression loading were determined. Results suggest that matrix materials that fail by delamination have the lowest damage tolerance, and it is concluded that alternative matrix materials with transverse reinforcement to suppress the delamination failure mode and yield the higher-strain value transverse shear crippling mode should be developed.

207. Rhodes, M. D.; Mikulas, M. M., Jr.; and McGowan, P. E.: Effect of Orthotropic Properties and Panel Width on the Compression Strength of Graphite-Epoxy Laminates With Holes. *23rd AIAA Structures, Structural Dynamics and Materials Conference* (New Orleans, La., May 10–12), *Collection of Technical Papers, Part 1,* 1982, pp. 338–351. (Available as AIAA Paper 82-0749.)

82A30112#

An experimental study conducted to evaluate the effect of laminate orthotropic properties and panel width on the compression strength of 48-ply graphite-epoxy laminates with drilled holes is described. The test results are evaluated on the basis of hole size and specimen width and are used in determining parameters necessary for predicting trends using the point stress failure criterion. Good agreement is obtained between experimental and predicted values of failure for panels fabricated from two quasi-isotropic laminates and one orthotropic laminate. The results suggest that panels of different widths having holes that are large in relation to the failure prediction parameter should be included in any test program conducted to develop prediction trends that may be used in design applications.

208. Sankar, B. V.; and Sun, C. T.: Low-Velocity Impact Response of Laminated Beams Subjected to Initial Stresses. *AIAA Journal,* vol. 23, Dec. 1985, pp. 1962–1969.

86A17146#

Finite element procedures are used in conjunction with a numerical algorithm to compute the impact response of a graphite-epoxy laminated beam subjected to tensile initial stresses. The effects of initial stresses on the contact duration, impact force, coefficient of restitution, and bending and shear stresses are discussed. The analytically computed contact force history and strain response are compared with some experimental results.

209. Schramm, S. W.; and Daniel, I. M.: *Embedded Gage Impact Study.* NASA CR-172227, Jan. 1984.

84N17558#

Impact damage in graphite/epoxy laminates was characterized and transient strain history during impact was correlated. The specimens were circular plates 12.7 cm (5 in.) in diameter and clamped along their circumference. The specimens were impacted with a 185 gm impactor, dropped from heights of 1.20 m and 1.65 m. An accelerometer was attached to the back surface of the specimen opposite the impact point and was used to trigger the recording instrumentation. The transient strain data were recorded with an eight channel waveform digitizer capable of sampling data at 0.5 μsec intervals. The data were stored, processed, and plotted by means of a microcomputer. Transient strain data were correlated with results from ultrasonic inspection of the specimens.

210. Sharma, A. V.: Low-Velocity Impact Tests on Fibrous Composite Sandwich Structures. *Test Methods and Design Allowables for Fibrous Composites* (Proceedings of the Symposium, Dearborn, Mich., Oct. 2–3, 1979), ASTM, 1981, pp. 54–70.

81A47805

The effect of low-velocity projectile impact on the load-carrying ability of the composite sandwich structural components is investigated experimentally. The impact simulates the damage caused by runway debris and the accidental dropping of hand tools during servicing on secondary aircraft structures made with composites.

211. Sharma, A. V.: *Design and Testing of Small Composite Specimens.* NASA CR-169059, Dec. 1981.

82N26393#

The effect of specimen size on the buckling strains of laminates subjected to low-velocity projectile impact was investigated. The fiber composite selected was the T300/5208 graphite/epoxy system. The quasi-isotropic laminates tested had 16 and 32 plies.

The results were compared with those of a 48-ply laminate. Specimens of three different lengths with length to width aspect ratios of 1, 1.5, and 2 were also studied. The results show that (1) the specimen length does not have any significant influence on the buckling strains at failure caused by the projectile impact, and (2) the influence of specimen thickness on the strains at failure decreases as the velocity of the impacting projectile increases.

212. Sharma, A. V.: Effect of Temperature on Composite Sandwich Structures Subjected to Low-Velocity Projectile Impact. ASME Paper 78-WA/AERO-2, 1978.

79A19716#

An experimental investigation was conducted to study the effect of low-velocity projectile impact on graphite/epoxy and Kevlar 49/epoxy sandwich structural components. Testing was performed at moderately low and high temperatures to assess the strength degradation of composites as compared to room temperature values. Low energy projectile impact is considered to simulate the damage caused by runway debris such as small rocks, dropping of hand tools during servicing, etc., on secondary aircraft structures fabricated out of composites. The preload and impact energy necessary to cause catastrophic failure were determined. The residual strength of impact-damaged specimens was also measured.

213. Sharma, A. V.: Damage Tolerance of Composite Sandwich Structures Subjected to Projectile Impact. *New Horizons—Materials and Processes for the Eighties* (Proceedings of the Eleventh National Conference, Boston, Mass., Nov. 13–15), Society for the Advancement of Material and Process Engineering, 1979, pp. 900–917.

80A34812

An experimental investigation was conducted to evaluate the effect of low-velocity projectile impact on the strength carrying ability of secondary aerospace structural components fabricated with graphite/epoxy composite materials. The preload and the impact energy combinations necessary to cause catastrophic failure were determined. Those specimens that survived the projectile impact were evaluated for residual strength.

214. Shivakumar, K. N.; Elber, W.; and Illg, W.: Prediction of Impact Force and Duration Due to Low-Velocity Impact on Circular Composite Laminates. *ASME Transactions, Journal of Applied Mechanics,* vol. 52, Sept. 1985, pp. 674–680.

86A10514#

Two simple and improved models—energy-balance and spring-mass—were developed to calculate impact force and duration during low-velocity impact of circular composite plates. Both models include the contact deformation of the plate and the impactor as well as bending, transverse shear, and membrane deformations of the plate. The plate was transversely isotropic graphite/epoxy composite laminate and the impactor was a steel sphere. Calculated impact forces from the two analyses agreed with each other. The analyses were verified by comparing the results with reported test data.

215. Shivakumar, K. N.; and Elber, W.: Delamination Growth Analysis in Quasi-Isotropic Laminates Under Loads Simulating Low-Velocity Impact. AIAA Paper 84-0962, 1984.

84A33141#

A geometrically nonlinear finite-element analysis has been developed to calculate the strain energy released by delaminating plates during impact loading. Only the first mode of deformation, which is equivalent to static deflection, was treated. Both the impact loading and delamination in the plate were assumed to be axisymmetric. The strain energy release rates in peeling and shear sliding modes were calculated using the fracture mechanics crack closure technique. Energy release rates for various delamination sizes and locations for various plate configurations and materials were compared.

216. Shivakumar, K. N.; Elber, W.; and Illg, W.: Analysis of Progressive Damage in Thin Circular Laminates Due to Static-Equivalent Impact Loads. *24th AIAA Structures, Structural Dynamics and Materials Conference* (Lake Tahoe, Nev., May 2–4), *Collection of Technical Papers, Part 2,* 1983, pp. 606–615. (Available as AIAA Paper 83-0997.)

83A29872#

Clamped circular graphite/epoxy plates (25.4-, 38.1-, and 50.8-mm radii) with an 8-ply quasi-isotropic layup were analyzed for static-equivalent impact loads using the minimum-total-potential-energy method and the von Karman strain-displacement equations. A step-by-step incremental transverse displacement procedure was used to calculate plate load and ply stresses. The ply failure region was calculated using the Tsai-Wu criterion. The corresponding failure modes (splitting and fiber failure) were determined using the maximum stress criteria. The first failure mode was splitting and initiated in the bottom ply. The splitting-failure thresholds were relatively low and tended to be lower for large

plates than for small plates. The splitting-damage region in each ply was elongated in its fiber direction; the bottom ply had the largest damage region. The calculated damage region for the 25.4-mm-radius plate agreed with limited static test results from the literature.

217. Shuart, M. J.; and Williams, J. G.: Compression Failure Characteristics of ±45-Deg-Dominated Laminates With a Circular Hole or Impact Damage. *25th AIAA Structures, Structural Dynamics and Materials Conference* (Palm Springs, Calif., May 14–16), *Technical Papers, Part 1,* 1984, pp. 399–409. (Available as AIAA Paper 84-0848.)

84A31669

An investigation of the compression failure characteristics of ±45-deg-dominated laminates with a circular hole or impact damage was conducted. Graphite-epoxy laminates consisting of all ±45-deg plies and of ±45-deg and 90-deg plies were evaluated.

218. Starnes, J. H. Jr.; Rhodes, M. D.; and Williams, J. G.: Effect of Impact Damage and Holes on the Compressive Strength of a Graphite/Epoxy Laminate. *Nondestructive Evaluation and Flaw Criticality for Composite Materials* (Proceedings of the Symposium, Philadelphia, Pa., Oct. 10–11, 1978), ASTM, 1979, pp. 145–171.

80A33385

An experimental investigation has been conducted to determine the effect of low-velocity impact damage and unloaded circular holes on the compressive strength of a 48-ply orthotropic graphite/epoxy flat laminate. Specimens were impacted by a 1.27-cm-diameter aluminum sphere with speeds from 52 to 101 m/s to simulate momenta typical of low-velocity impact hazards that can occur in commercial aircraft service. It is shown that low-velocity impact damage can significantly degrade the static compressive strength of the laminate. Specimens that fail at axial strains above 0.008 in the undamaged condition can fail at strains as low as 0.0031 when impacted at 100 m/s. Circular holes also reduce the static compressive strength of the laminate. The failure strain decreases as the hole diameter increases.

219. Sun, C. T.; and Chen, J. K.: On the Impact of Initially Stressed Composite Laminates. *Journal of Composite Materials,* vol. 19, Nov. 1985, pp. 490–504.

86A17637

The impact response behavior of initially stressed composite laminates is investigated using the finite element method. An experimentally established contact law is incorporated into the finite element program. The Newmark time integration algorithm is used for solving the time dependent equations of the plate and the impactor. Numerical results, including the contact force history, deflection, and strain in the plate, are presented. Effects of impact velocity, initial stress, and the mass and size of the impactor are discussed.

220. Sun, C. T.; and Sankar, B. V.: Smooth Indentation of an Initially Stressed Orthotropic Beam. *International Journal of Solids and Structures,* vol. 21, no. 2, 1985, pp. 161–176.

85A32568

The contact behavior between a smooth rigid cylinder and a simply supported orthotropic beam under uniaxial initial stresses is studied. The displacements are computed by superposing the Mindlin plate solution with the solution obtained from Biot's theory of incremental deformation. Finite Fourier transforms are used in solving the equations. A point matching technique is used to compute the contact stresses and the amount of indentation for a given contact length. The effects of orthotropy and initial stresses on the contact stress distribution are investigated. An indentation law is established from the numerical results.

221. Sykes, G. F.; and Stoakley, D. M.: Impact Penetration Studies of Graphite/Epoxy Laminates. *Materials 1980* (Proceedings of the Twelfth National Technical Conference, Seattle, Wash., Oct. 7–9), Society for the Advancement of Material and Process Engineering, 1980, pp. 482–493.

81A43632

A single epoxy system, NARMCO 5208, in both composite and cured neat-resin form was studied with a constant velocity impact test apparatus. The parameters investigated include resin cure temperature, fiber type, ply thickness and orientation, and impact velocity. The results from the study show that matrix chemistry, as obtained by cure temperature changes, has a significant effect upon the failure mode and energy absorption during impact.

222. Williams, J. G.; and Rhodes, M. D.: Effect of Resin on Impact Damage Tolerance of Graphite/Epoxy Laminates. *Composite Materials: Testing and Design (Sixth Conference),* ASTM, 1982, pp. 450–480.

84A27357

Twenty-four different epoxy resin systems were evaluated by a variety of test techniques to identify materials that exhibited improved impact damage tolerance in graphite/epoxy composite laminates. Forty-eight-ply composite panels of five of the material systems were able to sustain 100 m/s impact by a 1.27-cm-diameter aluminum projectile while statically loaded to strains of 0.005. Of the five materials with the highest tolerance to impact, two had elastomeric additives, two had thermoplastic additives, and one had a vinyl modifier; all the five systems used bisphenol A as the base resin. An evaluation of test results shows that the laminate damage tolerance is largely determined by the resin tensile properties, and that improvements in laminate damage tolerance are not necessarily made at the expense of room-temperature mechanical properties. The results also suggest that a resin volume fraction of 40 percent or greater may be required to permit the plastic flow between fibers necessary for improved damage tolerance.

223. Williams, J. G.; Anderson, M. S.; Rhodes, M. D.; Starnes, J. H., Jr.; and Stroud, W. J.: Recent Developments in the Design, Testing and Impact-Damage Tolerance of Stiffened Composite Panels. *Fibrous Composites in Structural Design,* Plenum Press, 1980, pp. 259–291.

82A27140

The structural technology of laminated filamentary-composite stiffened-panel structures under combined in-plane and lateral loadings is discussed. Emphasis is on analyzing the behavior of the structures under load, determining appropriate structural proportions for weight efficient configurations, and effects of impact damage and geometric imperfections on structural performance. Experimental data on buckling of panels under in-plane compression validate the analysis and sizing methods and illustrate structural performance and efficiency obtained from representative structures. It is shown that the strength of panels under in-plane compression can be degraded by low-velocity impact damage, and data are presented which indicate that the matrix is a significant factor influencing tolerance to impact damage.

224. Williams, J. G.: *Effect of Impact Damage and Open Holes on the Compression Strength of Tough Resin/High Strain Fiber Laminates—Graphite-Epoxy Composites.* NASA TM-85756, Feb. 1984.

84N20650#

Structural damage and design-based inclusions such as cutouts can reduce significantly the strength of graphite-epoxy laminates. One composite mechanics research activity at the Langley Research Center is to assess and improve the performance of composite structures. Reductions in strength are common to both tension and compression loaded laminates; however, the problem associated with compression performance is the most difficult to solve. Compression failure involves both shear crippling and delamination modes. Several graphite-epoxy material systems proposed for improved damage tolerance were studied. Material parameters included both tough resin formulations and high strain fibers.

225. Ying, L.: Role of Fiber/Matrix Interphase in Carbon Fiber-Epoxy Composite Impact Toughness. *SAMPE Quarterly*, vol. 14, Apr. 1983, pp. 26–33. (Presented at 38th Society of Plastics Industry Annual Conference of the Reinforced Plastics, Houston, Tex., Feb. 1983.)

84A14287

Experimental results were obtained which show the effect of the interphase on composite performance. Various finish variants were formulated, based on different chemical and mechanical properties and applied to Celion 6000 carbon fiber. Bond strength and failure mechanisms were studied.

Fatigue and fracture

226. Bakis, C. E.; and Stinchcomb, W. W.: Response of Thick, Notched Laminates Subjected to Tension-Compression Cyclic Loads. Paper presented at the ASTM Symposium on Composite Materials: Fatigue and Fracture, Dallas, Tex., Oct. 24–25, 1985.

86N21610#

The fatigue response of a T300-5208 graphite-epoxy laminate with a drilled center hole subjected to fully reversed tension-compression ($R = -1$) constant amplitude loading was investigated. Damage evaluation techniques such as stiffness monitoring, penetrant-enhanced X-ray radiography, C-scan, laminate deply, and residual strength were used to establish the mechanisms of damage development as well as the relations between this damage and the stiffness, strength, and life of the laminate.

227. Bigelow, C. A.: *Fatigue of Graphite/Epoxy Buffer Strip Panels With Center Cracks.* NASA TM-87595, Aug. 1985.

86N10581#

The effects of fatigue loading on the behavior of graphite/epoxy panels with either S-Glass or Kevlar-49 buffer strips are studied. Buffer strip panels were fatigued and tested in tension to measure their residual strength with crack-like damage. The buffer strips were parallel to the loading direction and made by replacing narrow strips of the 0-degree graphite plies with strips of either 0-degree S-Glass/epoxy or Kevlar-49/epoxy on a one-for-one basis. The panels were subjected to a fatigue loading spectrum MINITWIST, the shortened version of the standardized load program for the wing lower surface of a transport aircraft.

228. Bigelow, C. A.: *A Cracked Orthotropic Sheet Stiffened by a Semi-Infinite Orthotropic Sheet.* NASA TP-2455, July 1985.

85N32339#

The stress intensity factor is determined for a cracked orthotropic sheet adhesively bonded to an orthotropic stringer. Since the stringer is modeled as a semi-infinite sheet, the solution is most appropriate for a crack tip located near a stringer edge. Both adherends are treated as homogeneous, orthotropic media. It is assumed they are in plane stress and the adhesive is in pure shear.

229. Bigelow, C. A.: *Effect of Debond Growth on Stress-Intensity Factors in a Cracked Orthotropic Sheet Stiffened by a Semi-Infinite Orthotropic Sheet.* NASA TM-87598, Jan. 1986.

86N16614#

Stress-intensity factors are determined for a cracked infinite sheet adhesively bonded to a stringer, and debonding of the adhesive layer is predicted. The stringer is modeled as a semi-infinite sheet. Adhesive nonlinearity is also included. Both the sheet and stringer are treated as homogeneous, orthotropic materials.

230. Cantrell, J. H., Jr.; Winfree, W. P.; and Heyman, J. S.: Profiles of Fatigue Damage in Graphite/Epoxy Composites From Ultrasonic Transmission Power Spectra. *Recent Advances in Composites in the United States and Japan* (Proceedings of the Symposium, Hampton, Va., June 6–8, 1983), ASTM, 1985, pp. 197–206.

85A46535

Early fatigue damage in non-unidirectional, multi-ply graphite/epoxy composites is manifested by a distribution of cracks and disbonds through the bulk of the material. Such damage is subtle and is

difficult to detect with conventional ultrasonic technology. Consequently, a new ultrasonic measurement technique called phase-insensitive tone-burst spectroscopy has been developed. The new technique eliminates problems associated with phase cancellation and pulse shape artifacts inherent to conventional broadband ultrasonic spectral measurement systems and produces clean spectral information irrespective of specimen inhomogeneity or irregularities in surface geometry.

231. Crews, J. H., Jr.: Bolt-Bearing Fatigue of a Graphite/Epoxy Laminate. *Joining of Composite Materials* (Proceedings of the Symposium, Minneapolis, Minn., Apr. 16, 1980), ASTM, 1981, pp. 131–144.

82A20983

Graphite/epoxy laminates (T300/5208) were tested under bolt-bearing loads for a range of bolt clampup torques and for several test conditions involving water. High clampup torque improved both the static strength and fatigue limit by about 100 percent compared to a simple pin-bearing case, which had no through-the-thickness constraint. The static strength improvement was explained in terms of failure modes.

232. Dharani, L. R.; and Goree, J. G.: Analysis of a Hybrid, Unidirectional Buffer Strip Laminate. *Composite Structures 2* (Proceedings of the Second International Conference, Paisley, Scotland, Sept. 14–16), Applied Science Publishers (London), 1983, pp. 453–466.

84A10452

A method of analysis capable of predicting accurately the fracture behavior of a unidirectional composite laminate containing symmetrically placed buffer strips is presented. As an example, for a damaged graphite/epoxy laminate, the results demonstrate the manner in which to select the most efficient combination of buffer strip properties necessary to inhibit crack growth.

233. Dharani, L. R.; and Goree, J. G.: Analysis of a Unidirectional, Symmetric Buffer Strip Laminate With Damage. *Engineering Fracture Mechanics,* vol. 20, no. 5–6, 1984, pp. 801–811.

85A21928

A method for predicting the fracture behavior of hybrid buffer strip laminates is presented in which the classical shear-lag model is used to represent the shear stress distribution between adjacent fibers.

The method is demonstrated by applying it to a notched graphite/epoxy laminate, and the results show clearly the manner in which the most efficient combination of buffer strip properties can be selected in order to arrest the crack.

234. Francis, P. H.; Walrath, D. E.; Sims, D. F.; and Weed, D. N.: Biaxial Fatigue Loading of Notched Composites. *Journal of Composite Materials,* vol. 11, Oct. 1977, pp. 488–501.

78A17796#

Thin-walled, 2.54-cm diameter tubular specimens of graphite/epoxy were fatigue cycled in combinations of axial, torsional, and internal pressure loading. S-N curves were developed to characterize fatigue behavior under pure axial, torsional, or internal pressure loading, as well as combined loading fatigue.

235. Goree, J. G.; and Kaw, A. K.: *Shear-Lag Analysis of Notched Laminates With Interlaminar Debonding.* NASA CR-3798, May 1984.

84N23697#

The fracture behavior of a debonded zone of finite width with no longitudinal damage in the unidirectional ply is predicted and the solution is then extended to include longitudinal matrix yielding and splitting in the unidirectional ply at the crack tip. The shear-lag assumption is used to describe the shear transfer between fibers. The fracture behavior of the laminate is studied as a function of initial crack length, constraint ratio, and width of the debonded zone. Results indicate that debonding can reduce the maximum fiber stress at the crack tip on the order of ten percent.

236. Goree, J. G.; and Dharani, L. R.: Analysis of a Hybrid, Uni-Directional Laminate With Damage. *Mechanics of Composite Materials: Recent Advances* (Proceedings of the Symposium, Blacksburg, Va., Aug. 16–19, 1982), Pergamon Press, 1983, pp. 163–177.

84A33382

The fracture behavior of hybrid (buffer strip) laminated composites is predicted by a novel method in terms of material properties, geometry, and internal damage. Attention is given to the cases of broken fibers in a uni-directional half-plane, adjoined half-planes having different fiber and matrix properties, and the solution of two half-planes bounding a third distinct region of finite width. The analysis is based on a materials modeling approach using the classical shear-lag assumption to represent the stress transfer between fibers.

237. Goree, J. G.; Dharani, L. R.; and Jones, W. F.: *Mathematical Modeling of Damage in Unidirectional Composites.* NASA CR-3453, Aug. 1981.

81N29166#

A review of some approximate analytical models for damaged fiber reinforced composite materials is presented. Using the classical shear lag stress displacement assumption, solutions are presented for a unidirectional laminate containing a notch, a rectangular cut-out, and a circular hole. The models account for longitudinal matrix yielding and splitting as well as transverse matrix yielding and fiber breakage. The constraining influence of a cover sheet on the unidirectional laminate is also modeled.

238. Goree, J. G.: *Preliminary Investigation of Crack Arrest in Composite Laminates Containing Buffer Strips.* NASA CR-3000, 1978.

78N25459#

The mechanical properties of some hybrid buffer strip laminates and the crack arrest potential of laminates containing buffer strips were determined. The hybrid laminates consisted of graphite with either S-glass, E-glass, or Kevlar. Unnotched tensile coupons and center-cracked fracture coupons were tested. Elastic properties, complete stress/strain curves, and critical stress intensity values are given. The measured elastic properties compare well with those calculated by classical lamination theory for laminates with linear stress/strain behavior.

239. Goree, J. G.; and Wolla, J. M.: *Longitudinal Splitting in Unidirectional Composites, Analysis and Experiments.* NASA CR-3881, Apr. 1985.

85N22639#

An experimental study was conducted to determine the fracture behavior of center notched, unidirectional graphite/epoxy laminates when subjected to tensile loading. The actual behavior is compared to the behavior predicted by a mathematical model based on classical shear-lag assumptions.

240. Goree, J. G.: *The Accuracy of Approximate Solutions in the Analysis of Fracture of Composites.* NASA CR-175738, May 1985.

85N25437#

This paper concerns the accuracy of three related mathematical models (developed by Hedgepeth, Eringen and Sendeckyj, and Jones) used in the stress analysis and in fracture studies of continuous-fiber composites. These models have particular applications in the investigation of fiber and matrix stresses in unidirectional composites in the region near a crack tip. The interest in such models is motivated by the desire to be able to simplify the equations of elasticity to the point that they can be solved in a relatively easy manner.

241. Gregory, M. A.; Choksi, G. N.; and Herakovich, C. T.: Orthotropic Fracture Using a Singular Isoparametric Element. *Proceedings of the Pressure Vessels and Piping Conference* (New Orleans, La., June 23-26), ASME, 1985, pp. 227-234.

86A26895#

The six noded quarter point natural isoparametric triangular element is employed to obtain displacement and stress distributions in the vicinity of the crack tip in a center-cracked tensile coupon of unidirectional graphite epoxy. The material is considered to be homogeneous, elastic, and orthotropic. The finite element results are compared to the analytical solution of anisotropic elasticity.

242. Gregory, M. A.; and Herakovich, C. T.: Predicting Crack Growth Direction in Unidirectional Composites. *Journal of Composite Materials,* vol. 20, Jan. 1986, pp. 67-85.

86A23374

The purpose of this study was to gain a better understanding of the parameters affecting crack growth direction in unidirectional composite materials. To achieve this, the effect of anisotropy and biaxial far field loading on the direction of crack growth in unidirectional off-axis composite materials was investigated.

243. Grimes, G. C.: Structural Design Significance of Tension-Tension Fatigue Data on Composites. *Composite Materials: Testing and Design* (Proceedings of the Fourth Conference, Valley Forge, Pa., May 3-4, 1976), ASTM, 1977, pp. 106-119.

78A12058

Constant cycle tension-tension fatigue and related static tension data were generated on six single composite material/orientation combinations and twenty-one hybrid composite material/orientation combinations. The significance of these room temperature-dry data on the design allowables and weight of aerodynamic structures is discussed.

244. Hardrath, H. F.: Reliability Assurance of Aircraft Structures. *Proceedings of the Twenty-Fourth Sagamore Army Materials Research Conference* (Raquette Lake, N.Y., Aug. 21-26, 1977), Plenum Press, 1980, pp. 313-325.

80A48938

The procedures for assuring high reliability in aircraft structures are studied. Consideration is given to a statistical approach, airworthiness specifications, and their impact. Reliability of inspections and reliability in composite structures are outlined.

245. Harris, C. E.; Morris, D. H.; and Poe, C. C., Jr.: The Fracture Behavior of Filament Wound Cylinders With Surface Flaws. *26th AIAA Structures, Structural Dynamics and Materials Conference* (Orlando, Fla., Apr. 15–17), *Technical Papers, Part 1,* 1985, pp. 218–224. (Available as AIAA Paper 85-0650.)

85A30250#

The behavior of tensile coupons with surface notches of various semielliptical shapes has been evaluated for specimens obtained from a thick filament wound graphite/epoxy cylinder. Specimens with very shallow notches were observed to be notch insensitive. Specimens with deeper notches were sensitive to notch depth and notch aspect ratio.

246. Harris, C. E.; and Morris, D. H.: An Evaluation of the Effects of Stacking Sequence and Thickness on the Fatigue Life of Quasi-Isotropic Graphite/Epoxy Laminates. *Recent Advances in Composites in the United States and Japan* (Proceedings of the Symposium, Hampton, Va., June 6–8, 1983), ASTM, 1985, pp. 153–172.

85A46533

Notched and unnotched geometries at 16-, 32-, and 64-ply thicknesses of a 90/45/0/−45 (ns) laminate and a 45/0/−45/90 (ns) laminate were tested in compression-compression fatigue. The fatigue life and the initiation, type, and progression of damage were determined. Interlaminar stresses generated at straight, free edges of axially loaded laminates were used to interpret the test results. The fatigue lives of the notched specimens did not appear to be a strong function of laminate stacking sequence or specimen thickness.

247. Harris, C. E.; and Morris, D. H.: On the Use of Crack-Tip-Opening Displacement To Predict the Fracture Strength of Notched Graphite/Epoxy Laminates. *Experimental Mechanics,* vol. 25, June 1985, pp. 193–199. (Presented at 5th International Congress on Experimental Mechanics, Montreal, Canada, June 10–15, 1984.)

85A39235

The fracture strength of notched graphite/epoxy laminates was measured experimentally. Four replicate tests were conducted for a variety of laminate-stacking sequences, thicknesses, and notch lengths. Extensive notch-tip damage prior to fracture was characteristic of most laminates.

248. Harris, C. E.; and Morris, D. H.: A Comparison of Several Methods of Calculating Fracture Toughness From Composite Laminate Test Data. *Proceedings of the Society for Experimental Stress Analysis Spring Conference* (Cleveland, Ohio, May 15–19, 1983), 1984, pp. 405–412.

84A49592#

Critical fracture toughness was determined by two different techniques for graphite/epoxy laminates of three stacking sequences and several thicknesses. Critical fracture toughness was determined by a finite element stress analysis of a center-cracked tension specimen which yielded the stress intensity factor as a function of specimen thickness and applied load. As an alternative approach the critical strain energy release rate was determined from the compliance calibration technique. Test results from both procedures are compared and discussed.

249. Harris, C. E.; and Morris, D. H.: *Fracture Behavior of Thick, Laminated Graphite/Epoxy Composites.* NASA CR-3784, Mar. 1984.

84N20648

The research concentrated on the measurement of fracture toughness utilizing the center-cracked tension, compact tension, and three point bend specimen configurations. The development of subcritical damage at the crack tip was studied nondestructively using enhanced X-ray radiography and destructively using the laminate deply technique. The test results showed fracture toughness to be a function of laminate thickness.

250. Humphreys, E. A.; and Rosen, B. W.: *Development of a Realistic Stress Analysis for Fatigue Analysis of Notched Composite Laminates.* NASA CR-159119, May 1979.

80N11146#

A finite element stress analysis which consists of a membrane and interlaminar shear spring analysis was developed. This approach was utilized in order to model physically realistic failure mechanisms while maintaining a high degree of computational economy. The accuracy of the stress analysis predictions is verified through comparisons with other solutions to the

composite laminate edge effect problem. The stress analysis model was incorporated into an existing fatigue analysis methodology and the entire procedure was computerized. A fatigue analysis is performed upon a square laminated composite plate with a circular central hole. A complete description and users guide for the computer code FLAC (Fatigue of Laminated Composites) is included as an appendix.

251. Jones, D. L.; and Whitworth, H. A.: Determination of the Stiffness Reduction of Graphite/Epoxy Laminates Under Cyclic Loading. *25th AIAA Structures, Structural Dynamics and Materials Conference* (Palm Springs, Calif., May 14–16), *Technical Papers, Part 1,* 1984, pp. 428–435. (Available as AIAA Paper 84-0863.)

84A31672#

A series of tests has been performed to determine the effect of fatigue loading on the stiffness degradation of graphite/epoxy composite materials. Specimens were tested in tension-tension fatigue at a loading frequency of 10 Hz and a stress ratio of 0.1, for a wide range of stress levels. During this investigation, both static and dynamic stiffnesses were continually monitored for the range of stress levels tested, and the results are presented in tabular and graphical form.

252. Kennedy, J. M.: Fracture Behavior of Hybrid Composite Laminates. *24th AIAA Structures, Structural Dynamics and Materials Conference* (Lake Tahoe, Nev., May 2–4), *Collection of Technical Papers, Part 1,* 1983, pp. 68–78. (Available as AIAA Paper 83-0804.)

83A29736#

The tensile fracture behavior of 15 center-notched hybrid laminates was studied. Three basic laminate groups were tested: (1) a baseline group with graphite/epoxy plies, (2) a group with the same stacking sequence but where the zero-deg plies were one or two plies of S-glass or Kevlar, and (3) a group with graphite plies but where the zero-deg plies were sandwiched between layers of perforated Mylar. Results of the tests showed that the hybrid laminates had higher fracture toughnesses than comparable all-graphite laminates. The higher fracture toughness was due primarily to the larger damage region at the ends of the slit; delamination and splitting lowered the stress concentration in the primary load-carrying plies.

253. Kennedy, J. M.: Tensile Stress-Strain Behavior of Hybrid Composite Laminates. Paper presented at Society of Experimental Stress Analysis Spring Conference, Cleveland, Ohio, May 15–20, 1983.

83A28900#

A study was made of the stress-strain response of several hybrid laminates, and the damage was correlated with nonlinear stress-strain response and ultimate strength. The fibers used in the laminates were graphite, S-glass, and Kevlar. Some laminates with graphite fibers had perforated Mylar film between plies, which lowered the interlaminar bond strength. The laminate configurations were chosen to be like those of buffer strips in large panels and fracture coupons. Longitudinal and transverse specimens were loaded in tension to failure. Some specimens were radiographed to reveal damage due to edge effects. Stress-strain response is discussed in terms of damage shown by the radiographs. Ultimate strengths are compared with simple failure criteria, one of which accounts for damage.

254. Kress, G. R.; and Stinchcomb, W. W.: Fatigue Response of Notched Graphite/Epoxy Laminates. *Recent Advances in Composites in the United States and Japan* (Proceedings of the Symposium, Hampton, Va., June 6–8, 1983), ASTM, 1985, pp. 173–196.

85A46534

Tests were performed to determine the damage states caused by cyclic tensile loading in quasi-isotropic graphite/epoxy laminates with center holes. The influence of the stacking sequence on the initiation and interaction of damage modes and the relationship between damage, strength, stiffness, and life of the laminates were also studied.

255. Kulkarni, S. V.; and McLaughlin, P. V., Jr.: An Engineering Approach to the Prediction of Fatigue Behavior of Unnotched/Notched Fiber Reinforced Composite Laminates. *Proceedings of the 33rd Reinforced Plastics/Composites Institute Annual Conference* (Washington, D.C., Feb. 7–10), Society of the Plastics Industry, Inc., 1978, pp. 12-D1–12-D6.

79A15531#

An engineering approach is proposed for predicting unnotched/notched laminate fatigue behavior from basic lamina fatigue data. The fatigue analysis procedure was used to determine the laminate property (strength/stiffness) degradation as a function of fatigue cycles in uniaxial tension and in plane shear. These properties were then introduced into the failure model for a notched laminate to obtain

damage growth, residual strength, and failure mode. The approach is thus essentially a combination of the cumulative damage accumulation (akin to the Miner-Palmgren hypothesis and its derivatives) and the damage growth rate (similar to the fracture mechanics approach) philosophies. An analysis/experiment correlation appears to confirm the basic postulates of material wearout and the predictability of laminate fatigue properties from lamina fatigue data.

256. Liber, T.; and Daniel, I. M.: *Evaluation of Composite Flattened Tubular Specimens.* NASA CR-145353, June 1978.

78N25133#

Flattened tubular specimens of graphite/epoxy, S-glass/epoxy, Kevlar-49/epoxy, and graphite/S-glass/epoxy hybrid materials were evaluated under static and cyclic uniaxial tensile loading and compared directly with flat coupon data of the same materials generated under corresponding loading conditions. Statically tested graphite/epoxy, S-glass/epoxy, and Kevlar-49/epoxy flattened tube specimens exhibit somewhat higher average strengths than their corresponding flat coupons. Fatigue tested flattened tube specimens failed in parasitic modes resulting in lower fatigue strengths than the corresponding flat coupons.

257. Liber, T.; and Daniel, I. M.: *Evaluation of Hybrid Composite Materials in Cylindrical Specimen Geometries.* NASA CR-145006, Mar. 1976.

76N27361#

Static and fatigue properties of three composite materials and hybrids were examined. The materials investigated were graphite/epoxy, S-glass/epoxy, PRD-49 (Kevlar 49)/epoxy, and hybrids in angle-ply configurations. A new type of edgeless cylindrical specimen was developed. It is a flattened tube with two flat sides connected by curved sections and it is handled much like the standard flat coupon. Special specimen fabrication, tabbing, and tab region reinforcing techniques were developed. Axial modulus, Poisson's ratio, strength, and ultimate strain were obtained under static loading from flattened tube specimens of nine laminate configurations.

258. Liebowitz, H.; and Jones, D. L.: *Biaxial Tests of Flat Graphite/Epoxy Laminates.* NASA CR-165793, Oct. 1981.

82N12141#

The influence of biaxially applied loads on the strength of composite materials containing holes was analyzed. The analysis was performed through the development of a three-dimensional finite element computer program that is capable of evaluating fiber breakage, delamination, and matrix failure. Realistic failure criteria were established for each of the failure modes, and the influence of biaxial loading on damage accumulation under monotonically increasing loading was examined in detail. Both static and fatigue testing of specially designed biaxial specimens containing central holes were performed. Static tests were performed to obtain an understanding of the influence of biaxial loads on the fracture strength of composite materials and to provide correlation with the analytical predictions. The predicted distributions and types of damage are in reasonable agreement with the experimental results. A number of fatigue tests were performed to determine the influence of cyclic biaxial loads on the fatigue life and residual strength of several composite laminates.

259. Lindenmeyer, P. H.: *A Proposed Approach to the Application of Nonlinear Irreversible Thermodynamics to Fracture in Composite Materials.* NASA CR-166074, Apr. 1983.

83N22324#

The fracture criteria upon which most fracture mechanics are based involve an energy balance that is not appropriate for the fracture mechanics of viscoelastic materials such as polymer matrix composites. A more appropriate criterion based upon nonequilibrium thermodynamics and involving a power balance rather than an energy balance is proposed. This criterion is based upon a reformulation of the second law of thermodynamics which focuses attention on the total Legendre transform of energy expressed as a functional over time and space.

260. McLaughlin, P. V., Jr.; Kulkarni, S. V.; Huang, S. N.; and Rosen, B. W.: *Fatigue of Notched Fiber Composite Laminates. Part 1: Analytical Model.* NASA CR-132747, Mar. 1975.

76N15274#

A description is given of a semi-empirical, deterministic analysis for prediction and correlation of fatigue crack growth, residual strength, and fatigue life time for fiber composite laminates containing notches (holes). The failure model used for the analysis is based upon composite heterogeneous behavior and experimentally observed failure modes under both static and fatigue loading. The analysis is consistent with the wearout philosophy. Axial cracking

and transverse cracking failure modes are treated together in the analysis.

261. Odom, E. M.; and Adams, D. F.: Stiffness Reductions During Tensile Fatigue Testing of Graphite/Epoxy Angle-Ply Laminates. *Recent Advances in Experimental Characterization of Composites* (Proceedings of the Fall Meeting, Salt Lake City, Utah, Nov. 6–10), Society for Experimental Stress Analysis, 1983, pp. 1–5.

85A30152

The monitoring of modulus decay during fatigue testing as an indicator of damage accumulation and incipient failure has frequently been postulated. The present study was designed to generate tensile fatigue data under carefully controlled test conditions. A computerized data acquisition system was used to permit the measurement of dynamic modulus without interrupting the fatigue cycling. Two different 8-ply laminate configurations of a T300/5208 graphite/epoxy composite were tested.

262. Phillips, E. P.: Effects of Truncation of a Predominantly Compression Load Spectrum on the Life of a Notched Graphite/Epoxy Laminate. *Fatigue of Fibrous Composite Materials* (Proceedings of the Symposium, San Francisco, Calif., May 22–23, 1979), ASTM, 1981, pp. 197–212.

81A37137

Constant amplitude and transport wing spectrum compressive loading tests were used to explore the fatigue behavior of a notched, graphite/epoxy T300/5208 laminate specimen. Results indicate that (1) the amount of buckling near the notch allowed in the tests significantly affected fatigue life; (2) load spectrum truncation at either high or low ends produced lives greater than those obtained for the baseline complete-spectrum test, but with greater impact at the high-load end; and (3) the predictions of the Palmgren-Miner cumulative damage theory were found to always be far longer than those obtained in the spectrum loading tests.

263. Poe, C. C., Jr.: *A Single Fracture Toughness Parameter for Fibrous Composite Laminates.* NASA TM-81911, Mar. 1981.

81N22418#

A general fracture toughness parameter Q_c was derived and verified to be a material constant, independent of layup, for centrally cracked boron aluminum composite specimens. A limited amount of data indicated that the ratio Q_c/ϵ_{tuf}, where ϵ_{tuf} is

the ultimate tensile strain of the fibers, might be a constant for all composite laminates, regardless of material and layup. In that case, a single value Q_c/ϵ_{tuf} could be used to predict the fracture toughness of all fibrous composite laminates from only the elastic constants and ϵ_{tuf}. Values of Q_c/ϵ_{tuf} were calculated for centrally cracked specimens made from graphite/polyimide, graphite/epoxy, E glass/epoxy, boron/epoxy, and S glass graphite/epoxy materials with numerous layups. Within ordinary scatter, the data indicate that Q_c/ϵ_{tuf} is a constant for all laminates that did not split extensively at the crack tips or have other deviate failure modes.

264. Poe, C. C., Jr.: A Unifying Strain Criterion for Fracture of Fibrous Composite Laminates. *Engineering Fracture Mechanics,* vol. 17, no. 2, 1983, pp. 153–171.

83A13340

Sova (1980) has derived a general fracture toughness parameter (Q_c), which is a material constant. It defines the critical level of strains in the principal load-carrying plies. The present investigation is concerned with the calculation of values for the ratio of Q_c and the ultimate tensile strain of the fibers. The obtained data indicate that this ratio is reasonably constant for layups which fail largely by self-similar crack extension.

265. Poe, C. C., Jr.: *Fracture Toughness of Fibrous Composite Materials.* NASA TP-2370, Nov. 1984.

85N11136#

Laminates with various proportions of 0-deg, 45-deg, and 90-deg plies were fabricated from T300/5208 and T300/BP-907 graphite/epoxy prepreg tape material. The fracture toughness of each laminate orientation or layup was determined by testing center-cracked specimens, and it was also predicted with the general fracture-toughness parameter. The predictions were good except when crack-tip splitting was large, at which time the toughness and strengths tended to be underpredicted.

266. Ramkumar, R. L.: *Fatigue Degradation in Compressively Loaded Composite Laminates— Graphite-Epoxy Composites.* NASA CR-165681, Apr. 1981.

81N23192#

The effect of imbedded delaminations on the compression fatigue behavior of quasi-isotropic, T300/5208, graphite/epoxy laminates was investigated. Teflon imbedments were introduced during panel

layup to create delaminations. Static and constant amplitude ($R = 0$, $\omega = 10$ Hz) fatigue tests were conducted. S-N data and half-life residual strength data were obtained.

267. Ramkumar, R. L.; Kulkarni, S. V.; and Pipes, R. B.: *Evaluation and Expansion of an Analytical Model for Fatigue of Notched Composite Laminates.* NASA CR-145308, Mar. 1978.

78N21219#

The analytical and experimental study performed to expand the existing static and fatigue failure analysis is described. The analytical effort extended the analysis to include interlaminar effects, while the experimental effort developed methods to obtain basic experimental data required as input to the analysis. The static failure analysis for notched laminates was modified to include interlaminar effects near the notch. Three-dimensional elastic and two-dimensional elastic-plastic finite element analyses were performed for some notched laminates.

268. Ramkumar, R. L.: Compression Fatigue Behavior of Composites in the Presence of Delaminations. *Damage in Composite Materials: Basic Mechanisms, Accumulation, Tolerance, and Characterization*, ASTM, 1982, pp. 184–210.

83A14562

Three different stacking sequences of a 64-ply layup were tested in an experimental investigation of the effect of idealized imbedded delaminations on the compression fatigue behavior of quasi-isotropic T300/5208 graphite-epoxy laminates. Post-failure examination of specimen cross sections revealed the absence of any fiber microbuckling, suggesting that the predominant failure mode was the propagation of imbedded delaminations in the tab region.

269. Reifsnider, K. L.: The Effect of Lamination-Induced Stresses on Fatigue Damage Development at Internal Flaws. *Composites Technology Review*, vol. 3, Spring 1981, pp. 17–20.

82A15627

The effects of stresses induced by the lamination of off-axis plies to 0-deg lamina on the development of damage during the fatigue loading of the 0-deg plies are discussed. The transverse normal stresses in the plane of the laminae and the laminate created by the laminating constraints when an axial force is applied to the laminate are calculated in terms of a differential Poisson ratio between the ply in question in the unconstrained and constrained states.

270. Reifsnider, K. L.; and Highsmith, A.: The Relationship of Compliance Changes During Fatigue Loading to the Fracture of Composite Materials. *Advances in Fracture Research* (Proceedings of the Fifth International Conference on Fracture, Cannes, France, Mar. 29–Apr. 3, 1981), *Volume 1*, Pergamon Press, 1982, pp. 503–510.

83A21682

The study is based on measurements of the change in engineering stiffness values induced by the development of damage in composite laminates during quasi-static or cyclic (fatigue) loading. Results are presented for both notched and unnotched laminates. Compliance changes are found to be caused by damage events that bring about both global and local redistributions of stress. It is also found that the redistributions of stress determine the residual strength of the laminate. The quantitative link between compliance changes and fracture strength is the mechanics of the internal stress redistributions.

271. Reifsnider, K. L.; and Talug, A.: Analysis of Fatigue Damage in Composite Laminates. *International Journal of Fatigue*, vol. 2, Jan. 1980, pp. 3–11.

81A23913

An attempt is made to give a comprehensive characterization of degradation in laminated composites under cyclic (fatigue) loading, based on the discovery of a "characteristic damage state" that forms independently of load history and is determined by laminae properties, orientation, and stacking sequence alone. The detailed nature of this state, its formation, and its influence on strength, life, and stiffness is discussed.

272. Reifsnider, K. L.: Some Fundamental Aspects of the Fatigue and Fracture Response of Composite Materials. *Recent Advances in Engineering Science* (Proceedings of the Fourteenth Annual Meeting, Bethlehem, Pa., Nov. 14–16), Lehigh University, 1977, pp. 373–384.

78A40318

A simple analytical model, based on the concept that the constraining plies surrounding cracks through the thickness of a given ply or plies control the spacing of those cracks by controlling the surrounding stress state, appears to agree well with experimental data. The analytical model, which uses an equilibrium element approach in the present case, is sensitive to lamina and laminate elastic properties, flaw size, stacking sequence, and an interlaminar stress transfer distance which is estimated from

experimental observations. The analysis is easily extended to study flaw growth and plastic region formation.

273. Rosen, B. W.; Kulkarni, S. V.; and McLaughlin, P. V., Jr.: Failure and Fatigue Mechanisms in Composite Materials. *Inelastic Behavior of Composite Materials* (Proceedings of the Winter Annual Meeting, Houston, Tex., Nov. 30–Dec. 5), ASME, 1975, pp. 17–72.

76A22462

A phenomenological description of microfailure under monotonic and cyclic loading is presented, emphasizing the significance of material inhomogeneity for the analysis. Failure in unnotched unidirectional laminates is reviewed for the cases of tension, compression, shear, transverse normal, and combined loads. The failure of notched composite laminates is then studied, with particular attention paid to the effect of material heterogeneity on load concentration factors in circular holes in such laminates, and a "materials engineering" shear-lay type model is presented. The fatigue of notched composites is discussed with the application of a "mechanistic wearout" model for determining crack propagation as a function of the number of fatigue cycles.

274. Ryder, J. T.; and Crossman, F. W.: *A Study of Stiffness, Residual Strength and Fatigue Life Relationships for Composite Laminates*. NASA CR-172211, Oct. 1983.

84N29978#

A study was made of the relationship between stiffness, strength, fatigue life, residual strength, and damage of unnotched, graphite/epoxy laminates subjected to tension loading.

275. Salkind, M. J.: Early Detection of Fatigue Damage in Composite Materials. AIAA Paper 75-772, 1975.

75A32660#

Early detection of fatigue damage in composite materials by nondestructive inspection (NDI) techniques has been demonstrated for glass/epoxy, graphite/glass/epoxy, and graphite/epoxy composites. Modulus and temperature were monitored and a correlation between them observed.

276. Stinchcomb, W. W.; Reifsnider, K. L.; Yeung, P.; and Masters, J.: Effect of Ply Constraint on Fatigue Damage Development in Composite Material

Laminates. *Fatigue of Fibrous Composite Materials* (Proceedings of the Symposium, San Francisco, Calif., May 22–23, 1979), ASTM, 1981, pp. 64–84.

81A37130

The effects of constraint on the response of composite materials can be classified as (1) in-plane effects, and (2) through-the-thickness effects, with inplane constraint being the principal contributor to changes in strength under quasi-static loading. It is also determined that the constraint situations that produce the greatest static strength do not minimize the extent of damage that develops under either static or cyclic loading. It is concluded that the mode and the extent of damage in notched and unnotched constrained plies is governed by the stress state in those plies.

277. Stinchcomb, W. W.; Henneke, E. G.; Reifsnider, K. L.; and Kress, G. R.: *Fatigue Damage in Notched Composite Laminates Under Tension-Tension Cyclic Loads*. NASA CR-176412, Sept. 1985.

86N14318#

The results are given of an investigation to determine the damage states which develop in graphite epoxy laminates with center holes due to tension-tension cyclic loads, to determine the influence of stacking sequence on the initiation and interaction of damage modes and the process of damage development, and to establish the relationships between the damage states and the strength, stiffness, and life of the laminates. Two quasi-isotropic laminates were selected to give different distributions of interlaminar stresses around the hole. The laminates were tested under cyclic loads ($R = 0.1$, 10 Hz) at maximum stresses ranging between 60 and 95 percent of the notched tensile strength.

278. Sun, C. T.; and Chim, E. S.: Fatigue Retardation Due to Creep in a Fibrous Composite. *Fatigue of Fibrous Composite Materials* (Proceedings of the Symposium, San Francisco, Calif., May 22–23, 1979), ASTM, 1981, pp. 233–242.

81A37139

Experiments performed to determine the effects of temperature and creep strain on the fatigue life of a graphite/epoxy composite laminate show that (1) sustained periods of static loading have significant retardation effects on the fatigue degradation rate of the laminated composite, and (2) cyclic loading with a lower frequency preceding higher frequency loading also prolongs the fatigue life. A creep crack

hypothesis is proposed that can provide reasonable interpretations for these experimental results.

279. Sun, C. T.; and Chan, W. S.: Frequency Effects on the Fatigue Life of a Laminated Composite. *Composite Materials: Testing and Design* (Proceedings of the Fifth Conference, New Orleans, La., Mar. 20–22, 1978), ASTM, 1979, pp. 418–430.

80A21146

Effects of load frequency on the fatigue behavior of notched laminated graphite/epoxy composites are investigated. Experimental results are obtained using four frequencies at three load levels. Temperature rises near the center hole are also measured. A theoretical model based upon crack propagation in viscoelastic media is presented. The model is modified to account for the temperature effect.

280. Whitcomb, J. D.: Experimental and Analytical Study of Fatigue Damage in Notched Graphite/ Epoxy Laminates. *Fatigue of Fibrous Composite Materials* (Proceedings of the Symposium, San Francisco, Calif., May 22–23, 1979), ASTM, 1981, pp. 48–63.

81A37129

Fatigue damage development in notched graphite/ epoxy laminates was investigated in both compression and tension, with tested specimens being examined for damage type and location using light microscopy, scanning electron microscopy, ultrasonic C-scan, and X-radiography. Delamination and ply cracking were found to be the dominant types of fatigue damage. Comparison of observed delamination locations with finite element calculations indicates that both interlaminar shear and peel stresses must be considered when predicting delamination. Fatigue loading generally caused only small stiffness losses, and residual strengths were in all cases greater than, or equal to, the virgin strengths.

281. Whitcomb, J. D.: *Experimental and Analytical Study of Fatigue Damage in Notched Graphite/ Epoxy Laminates.* NASA TM-80121, June 1979.

79N27245#

Both tension and compression fatigue behaviors were investigated in four notched graphite/epoxy laminates. After fatigue loading, specimens were examined for damage type and location using visual inspection, light microscopy, scanning electron microscopy, ultrasonic C-scans, and X-radiography. Delamination and ply cracking were found to be the dominant types of fatigue damage. In general, ply

cracks did not propagate into adjacent plies of differing fiber orientation. To help understand the varied fatigue observations, the interlaminar stress distribution was calculated with finite element analysis for the regions around the hole and along the straight free edge. Comparison of observed delamination locations with the calculated stresses indicated that both interlaminar shear and peel stresses must be considered when predicting delamination.

282. Yang, J. N.; and Jones, D. L: Load Sequence Effects on Graphite/Epoxy Laminates. *Long-Term Behavior of Composites* (Proceedings of the Symposium, Williamsburg, Va., March 9–10, 1982), ASTM, 1983, pp. 246–262.

84A24890

Load sequence effects on composite material fatigue behavior are due to the difference in residual strength when fatigue occurs, or the "boundary effect," and the "material memory effect" due to previous loads. The present model, which allows the isolation of the effects of memory from those of boundary, has been investigated through a test program for graphite/epoxy laminates under high-low and low-high load sequences which generated statistical residual strength data. As predicted by the model, the memory effect is present.

283. Yang, J. N.; and Jones, D. L.: Fatigue of Graphite/Epoxy /0/90/45/−45/s Laminates Under Dual Stress Levels. *Composites Technology Review*, vol. 4, Fall 1982, pp. 63–70.

83A12060

A model for the prediction of loading sequence effects on the satistical distribution of fatigue life and residual strength in composite materials is generalized and applied to (0/90/45/−45)s graphite/epoxy laminates. Load sequence effects are found to be caused by both the difference in residual strength when failure occurs (boundary effect) and the effect of previously applied loads (memory effect). The model allows isolation of these two effects.

284. Yang, J. N.; and Jones, D. L.: Statistical Fatigue of Unnotched Composite Laminates. *Advances in Composite Materials* (Proceedings of the Third International Conference on Composite Materials, Paris, France, Aug. 26–29), *Volume 1*, Pergamon Press, 1980, pp. 472–483.

81A40532

A general analytical, phenomenological model is developed for characterizing fatigue damage accumulation in unnotched composite laminates, and experimental results are used to evaluate the model. The fatigue model, which is developed for various types of cyclic loadings, is based on the assumption that the residual strength reduction is a monotonically decreasing function of the fatigue life. Parameters are determined from baseline test data consisting of one set of residual strength data, one set of static strength data, and one set of constant amplitude fatigue life data.

285. Yang, J. N.; and Jones, D. L.: Load Sequence Effects on the Fatigue of Unnotched Composite Materials. *Fatigue of Fibrous Materials* (Proceedings of the Symposium, San Francisco, Calif., May 22–23, 1979), ASTM, 1981, pp. 213–232.

81A37138

A comprehensive version of an earlier fatigue and residual strength degradation model is proposed to predict the effect of load sequence on the statistical fatigue behavior of composite laminates.

286. Yang, J. N.; and Sun, C. T.: Proof Test and Fatigue of Unnotched Composite Laminates. *Journal of Composite Materials*, vol. 14, Apr. 1980, pp. 168–176.

80A37173

A comprehensive fatigue and residual strength degradation model has been used to predict the effect of proof loads (or high loads) on the statistical fatigue behavior of composite laminates. The validity of the theoretical model is confirmed by the experimental test results. The correlation between the test results and the theoretical distributions of the fatigue life and the residual strength for composite specimens with or without the effect of proof loads is shown to be very good.

287. Yang, J. N.; and Jones, D. L.: Statistical Fatigue of Graphite/Epoxy Angle-Ply Laminates in Shear. *Journal of Composite Materials*, vol. 12, Oct. 1978, pp. 371–389.

79A15483

A three-parameter fatigue and residual strength degradation model has been proposed to predict statistically the fatigue behavior of composite laminae under axial shear loadings. The fatigue behavior includes the fatigue life and the fatigue damage expressed in terms of the residual strength degradation.

It is shown that the correlation between the theoretical predictions and the test results on the statistical distributions of the fatigue life and the residual strength is excellent.

288. Yang, J. N.; and Jones, D. L.: The Effect of Load Sequence on the Statistical Fatigue of Composites. *20th AIAA Structures, Structural Dynamics and Materials Conference* (St. Louis, Mo., Apr. 4–6), *Technical Papers on Dynamics and Loads*, 1979, pp. 232–239. (Available as AIAA Paper 79-0760.)

79A28274#

A theoretical model to predict the effect of loading sequence on the statistical distributions of the fatigue life and the residual strength under n-stress levels of cyclic loading is derived on the basis of a three-parameter model. In particular, the dual stress fatigue cumulative damage is studied, showing that Miner's sum (the cumulative damage sum at fatigue failure) is a statistical variable. It is proved theoretically that Miner's sum is always greater than or equal to unity for the high-low load sequence, while it is always smaller than or equal to unity for the low-high load sequence, with the deviation from unity increasing as the difference between the high and low stress levels increases.

289. Yeung, P. C.; Stinchcomb, W. W.; and Reifsnider, K. L.: Characterization of Constraint Effects on Flaw Growth—In Graphite-Epoxy Composite Materials. *Nondestructive Evaluation and Flaw Criticality for Composite Materials* (Proceedings of the Symposium, Philadelphia, Pa., Oct. 10–11, 1978), ASTM, 1979, pp. 316–338.

80A33392

Experimental results are presented for the case of a flawed unidirectional lamina constrained by off-axis unflawed plies under static and fatigue loading. Flaw growth in the interior of the laminate, around the embedded flaw, was followed by nondestructive testing methods including video-thermographs and ultrasonic C-scan schemes.

Failure and delamination mechanics

290. Adams, D. S.; Bowles, D. E.; and Herakovich, C. T.: *Characteristics of Thermally-Induced Transverse Cracks in Graphite-Epoxy Composite Laminates*. NASA TM-85429, June 1983.

83N36109#

The characteristics of thermally-induced transverse cracks in T300/5208 graphite-epoxy cross-ply and quasi-isotropic laminates were investigated both experimentally and analytically. The formation of transverse cracks and the subsequent crack spacing present during cool down to $-250°F$ (116K) and thermal cycling between 250 and $-250°F$ (116 and 394K) was investigated. The state of stress in the vicinity of a transverse crack and the influence of transverse cracking on the laminate coefficient of thermal expansion (CTE) was predicted using a generalized plane strain finite element analysis and a modified shear lag analysis. A majority of the cross-ply laminates experienced transverse cracking during the initial cool down to $-250°F$, whereas the quasi-isotropic laminates remained uncracked.

291. Arin, K.: An Orthotropic Laminate Composite Containing a Layer With a Crack. *International Journal of Engineering Science*, vol. 15, no. 9-10, 1977, pp. 545-559.

A laminate composite containing an orthotropic layer with a crack situated normal to the interfaces and bonded to two orthotropic half-planes of dissimilar materials is considered. The solutions for two different classes of orthotropic materials are presented separately. In each case, the problem is first reduced to a system of dual integral equations, then to a singular integral equation which is subsequently solved numerically for the stress intensity factors at the tip of the crack. The effect of the material properties on the stress intensity factor is investigated. Two cases, the generalized plane stress and the plane strain, are treated simultaneously.

292. Babcock, C. D.; and Waas, A. M.: *Effect of Stress Concentrations in Composite Structures.* NASA CR-176410, Nov. 1985.

An experimental program is being carried out to gain a fundamental understanding of the failure mechanics of multilayered composite structures. As a part of this continuing study, the performance of laminated composite plates in the presence of a stress gradient and the failure of composite structures at points of thickness discontinuity is assessed. In particular, the questions of initiation of failure and its subsequent growth to complete failure of the structure are addressed.

293. Bauld, N. R., Jr.; Goree, J. G.; and Tzeng, L.-S.: A Comparison of Finite-Difference and Finite-Element Methods for Calculating Free Edge Stresses in Composites. *Computers and Structures*, vol. 20, no. 5, 1985, pp. 897-914.

Various numerical methods have been utilized in attempts to calculate the interlaminar stress components which precede delamination in a laminate. The present investigation has the objective to assess the capacity of a finite difference method to predict the character and magnitude of the interlaminar stress distributions near an interface corner. A second purpose of the investigation is to determine if predictions by the finite element method of in-plane, interlaminar stress components near an interface corner represent actual laminate behavior.

294. Bostaph, G. M.; and Elber, W.: A Fracture Mechanics Analysis for Delamination Growth During Impact on Composite Plates. *1983 Advances in Aerospace Structures, Materials and Dynamics* (Proceedings of the Symposium on Composites, Boston, Mass., Nov. 13-18), ASME, 1983, pp. 133-138.

A fracture mechanics analysis has been developed that describes the progress of delamination damage in composite plates struck by a hard spherical object. The analysis is based on large deflection plate mechanics for circular isotropic plates wherein multiple axisymmetric delaminations grow. Test data show that the analysis predicts the influence of plate thickness, support conditions, and matrix toughness on the onset and propagation of delamination.

295. Bowles, D. E.: Effect of Microcracks on the Thermal Expansion of Composite Laminates. *Journal of Composite Materials*, vol. 18, Mar. 1984, pp. 173-187.

A finite element analysis was used to quantitatively predict the effect of matrix microcracks in the 90 deg plies of graphite/epoxy composites on the coefficient of thermal expansion in the 0 deg direction.

296. Buczek, M. B.; and Herakovich, C. T.: Direction of Crack Growth in Fibrous Composites. *Mechanics of Composite Materials—1983* (Proceedings of the Symposium, Boston, Mass., Nov. 13-18), ASME, 1983, pp. 75-82.

A model for predicting the direction of crack growth in resin matrix composite materials is presented. The model is based upon the normal stress and the anisotropic tensile strength on arbitrary planes about the tip of a crack. The model is incorporated into a finite element solution which predicts energy release rates in addition to the direction of crack growth. Results are obtained for cracks in unidirectional off-axis slotted tensile coupons. Comparisons are made with other theories and experimental results.

297. Buczek, M. B.; and Herakovich, C. T.: *Finite Element Models for Predicting Crack Growth Characteristics in Composite Materials.* NASA CR-169478, Oct. 1982.

83N12213#

Two dimensional and quasi-three dimensional, linear elastic finite element models for the prediction of crack growth characteristics, including crack growth direction, in laminated composite materials are presented. The modified crack closure method is used to predict the applied load level for crack extension, and two failure theories, modifications of the point stress and the Hashin failure criteria, are proposed to predict the direction of crack extension in composites.

298. Buczek, M. B.; and Herakovich, C. T.: A Normal Stress Criterion for Crack Extension Direction in, Orthotropic Composite Materials. *Journal of Composite Materials*, vol. 19, Nov. 1985, pp. 544-553.

86A17641

A criterion for predicting the direction of crack extension in orthotropic composite materials is presented. The criterion is based upon the normal stress and the anisotropic tensile strength on arbitrary planes about the tip of a crack.

299. Chai, H.: The Characterization of Mode I Delamination Failure in Non-Woven, Multidirectional Laminates. *Composites*, vol. 15, Oct. 1984, pp. 277-290.

85A15621#

The uniform double cantilever beam test and SEM are presently used for the characterization of Mode I delamination behavior in fiber-reinforced epoxy laminates. Delamination failure assumes forms that depend on ply orientation, test specimen geometry, and matrix toughness, but the calculated fracture energy is noted to be heavily dependent on fracture surface morphology. A material property

concept that is independent of both test specimen geometry and the orientation of the plies constituting the delaminating interface is elucidated, through the definition of interlaminar fracture solely in terms of an interlaminar separation that includes no fiber breakage or pull-out. This value, which dissipates the lowest possible amount of energy during crack growth, is the controlling factor for laminate toughness.

300. Chai, H.; and Babcock, C. D.: Two-Dimensional Modelling of Compressive Failure in Delaminated Laminates. *Journal of Composite Materials*, vol. 19, Jan. 1985, pp. 67-98.

85A30850

An analytical model is developed to assess the compressive strength criticality of near-surface interlaminar defects in laminated composites. The delaminated region is elliptic in shape, separating a thick isotropic plate from a thin orthotropic layer whose material axes coincide with the ellipse axes. The growth conditions and growth behavior of this defect are studied by breaking the overall problem into an elastic stability problem and a fracture problem.

301. Chai, H.; Babcock, C. D.; and Knauss, W. G.: One Dimensional Modelling of Failure in Laminated Plates by Delamination Buckling. *International Journal of Solids and Structures*, vol. 17, no. 11, 1981, pp. 1069-1083.

81A43022

When low speed objects impact composite laminated plates, delamination may result. Under in-plane compression such delaminations may buckle and tend to enlarge the delaminated area which can lead to loss of global plate stability. This process is modelled here in a first attempt by a delaminating beam-column wherein the local delamination growth, stability, and arrest are governed by a fracture mechanics-based energy release rate criterion.

302. Delale, F.: *Fracture of Composite Orthotropic Plates.* NASA CR-145063, Mar. 1976.

76N30298#

The fracture problem of laminated plates which consist of orthotropic layers is considered. Symmetrical cracks are located normal to the bimaterial interfaces. The external loads are applied away from the crack region. Three cases are considered: (a) the case of internal cracks; (b) the case of broken laminates; and (c) the case of a crack crossing the interface. A general formulation of the problem is given for plane

strain and generalized plane stress cases. The singular behavior of stresses at the crack tips and at the interfaces is studied. In each case the stress intensity factors are computed for various crack geometries.

303. Elber, W.: *Toughening of Graphite-Epoxy Composites by Interlaminar Perforated Mylar Films.* NASA TM-78643, Mar. 1978.

78N19303

Fracture and notch strength tests of graphite-epoxy composites showed that unidirectional layups generally exhibit longitudinal cracking before failure, whereas multidirectional layups fail transversely with little longitudinal cracking. A simple qualitative analysis suggested that the higher matrix shear stresses in unidirectional materials cause the longitudinal cracking, and that this cracking was responsible for the high toughness of unidirectional composites. In a series of comparative tests, the interlaminar strength of multidirectional composites was reduced by placing perforated Mylar films between laminae; tests on notched and slotted specimens showed that the interlaminar films promoted delamination and longitudinal cracking near the notches and that, as a result, toughness, notch strength, and impact strength were substantially increased.

304. Gillespie, J. W., Jr.; Carlsson, L. A.; Pipes, R. B.; Rothschilds, R.; Trethewey, B.; and Smiley, A.: *Delamination Growth in Composite Materials.* NASA CR-178066, Feb. 1986.

86N21618#

The double cantilever beam (DCB) and the end notched flexure (ENF) specimens are employed to characterize MODE I and MODE II interlaminar fracture resistance of graphite/epoxy (CYCOM 982) and graphite/PEEK (APC2) composites. Sizing of test specimen geometries to achieve crack growth in the linear elastic regime is presented. Data reduction schemes based upon beam theory are derived for the ENF specimen and include the effects of shear deformation and friction between crack surfaces.

305. Hahn, H. T.; and Williams, J. G.: *Compression Failure Mechanisms in Unidirectional Composites.* NASA TM-85834, Aug. 1984.

84N33523#

Compression failure mechanisms in unidirectional composites were examined. Possible failure modes of constituent materials are summarized and analytical models for fiber microbuckling are reviewed from a unified viewpoint. Due to deficiencies in available models, a failure model based on nonlinear properties and initial fiber curvature is proposed. The effect of constituent properties on composite compression behavior was experimentally investigated using two different graphite fibers and four different epoxy resins. The predominant microscopic scale failure mode was found to be shear crippling.

306. Harris, C. E.; and Morris, D. H.: Role of Delamination and Damage Development on the Strength of Thick Notched Laminates. *Delamination and Debonding of Materials*, ASTM, 1985, pp. 424–447.

86A20646

A large, comprehensive program is being conducted at Virginia Tech to study the effect of laminate thickness on the fracture strength of notched laminated composites. Part of this program has been the study of the characteristics and development of subcritical crack-tip damage prior to failure. The study has concentrated on the center-cracked tension specimen geometry. Subcritical crack-tip damage has been studied using enhanced X-ray radiography and the laminate deply technique.

307. Harris, C. E.; and Morris, D. H.: A Damage Tolerant Design Parameter for Graphite/Epoxy Laminated Composites. *Journal of Composites Technology and Research*, vol. 7, Fall 1985, pp. 77–81.

86A13869

The results of an extensive experimental investigation of the fracture behavior of notched graphite/epoxy laminates are presented. Twenty-seven laminates with different stacking sequences and laminate thicknesses have been studied. The data presented herein suggest that a damage tolerant design parameter exists that is relatively independent of ply stacking sequence and laminate thickness.

308. Herakovich, C. T.; Gregory, M. A.; and Beuth, J. L., Jr.: *Crack Growth Direction in Unidirectional Off-Axis Graphite Epoxy.* NASA CR-175752, Dec. 1984.

85N25433#

An anisotropic elasticity crack tip stress analysis is implemented using three crack extension direction criteria (the normal stress ratio, the tensor polynomial, and the strain energy density) to predict the direction of crack extension in unidirectional off-axis graphite-epoxy. The theoretical predictions of crack extension direction are then compared with experimental results for 15 deg off-axis tensile coupons with

center cracks. Specimens of various aspect ratios and crack orientations are analyzed. It is shown that only the normal stress ratio criterion predicts the correct direction of crack growth.

309. Herakovich, C. T.: *Influence of Layer Thickness on the Strength of Angle-Ply Laminates*, NASA CR-168892, Apr. 1982.

82N23554#

Experimental results are presented showing that the strength and toughness of finite width angle-ply laminates can be increased significantly by using an alternating layer stacking sequence as opposed to a clustered configuration. The ultimate tensile stress of an alternating $\pm\theta$ laminate can be as much as 1.5 times that of a clustered configuration. Further, the toughness of the alternating layer configuration can be as much as 2.7 times that of the clustered configuration. These differences are explained analytically through consideration of the influence of layer thickness on the magnitude of the interlaminar shear stress and by examination of failed specimens.

310. Herakovich, C. T.; Nagarkar, A.; and O'Brien, D. A.: Failure Analysis of Composite Laminates With Free Edges. *Proceedings of the Winter Annual Meeting* (New York, N.Y., Dec. 2–7), ASME, 1979, pp. 53–66.

80A27983#

Linear elastic stress distributions obtained from a refined finite element mesh are used in conjunction with the tensor polynomial failure criterion to predict the initiation of failure in symmetric, finite-width graphite-epoxy laminates under tensile loading. Results are presented for a wide variety of laminates.

311. Herakovich, C. T.: On the Relationship Between Engineering Properties and Delamination of Composite Materials. *Journal of Composite Materials*, vol. 15, July 1981, pp. 336–348.

82A11547

Delamination of composite materials has been investigated with emphasis on the relationship between the engineering properties of the individual layers and edge effects. It is shown that interlaminar shear stresses are primarily a function of the mismatch in coefficients of mutual influence which can be as much as ten times greater than the mismatch in Poisson's ratio. The mismatch in coefficients of mutual influence has a high peak value in the 10–15 deg range for $\pm\theta$ laminates (where θ is the angle of fiber orientation measured from the axis of the coupon). This mismatch is reduced by a factor of two when the $\pm\theta$ layers are interspersed between 0 and 90 deg layers. Application of the results to composite design is illustrated by an example.

312. Herakovich, C. T.: On Failure Modes in Finite Width Angle Ply Laminates. *Proceedings of the Third International Conference on Composite Materials* (Paris, France, Aug. 26–29), *Volume 1*, Pergamon Press, 1980, pp. 425–435.

81A40528

The results of a failure analysis of finite width $(\pm\theta)$s T300/5208 graphite-epoxy laminates subjected to thermal curing loads and applied tensile strain are presented. Stress distributions obtained from linear elastic and linear thermoelastic finite element stress analyses are used in conjunction with the tensor polynomial failure criterion to predict the mode and location of first failure in laminates with fiber angles ranging from 5° to 75°. It is shown that the mode of failure is predominantly interlaminar shear for small fiber angles, shifts to combined interlaminar normal, in-plane shear and in-plane normal for intermediate angles, and is primarily transverse tension for large fiber angles.

313. Herakovich, C. T.; Post, D.; Buczek, M. B.; and Czarnek, R.: *Free Edge Strain Concentrations in Real Composite Laminates: Experimental-Theoretical Correlation.* NASA CR-173818, June 1984.

84N28919

The magnitude of the maximum shear strain at the free edge of axially loaded composite laminates was investigated experimentally and numerically to ascertain the actual value of strain concentration in resin matrix laminates and to determine the accuracy of finite element results. Experimental results using Moiré interferometry show large, but finite, shear strain concentrations at the free edge of graphite-epoxy and graphite-polyimide laminates. Comparison of the experimental results with those obtained using several different finite element representations showed that a four node isoparametric finite element provided the best and most trouble free numerical results.

314. Johnson, W. S., ed.: *Delamination and Debonding of Materials.* ASTM Special Technical Publication No. 876, 1985.

86A20626

The general topics consist of stress analysis, mechanical behavior, and fractography/NDI of composite laminates. Papers are presented on a dynamic hybrid finite-element analysis for interfacial cracks in composites, energy release rate during delamination crack growth in composite laminates, matrix deformation and fracture in graphite-reinforced epoxies, and the role of delamination and damage development on the strength of thick notched laminates. In addition, consideration is given to a new ply model for interlaminar stress analysis, a fracture mechanics approach for designing adhesively bonded joints, the analysis of local delaminations and their influence on composite laminate behavior, and moisture and temperature effects on the mixed-mode delamination fracture of unidirectional graphite/epoxy.

315. Kaw, A. K.; and Goree, J. G.: Shear-Lag Analysis of Notched Laminates With Interlaminar Debonding. *Engineering Fracture Mechanics*, vol. 22, no. 6, 1985, pp. 1013–1029.

86A18311

The fracture behavior of a debonded zone of finite width with no longitudinal damage in the unidirectional ply is predicted and the solution is then extended to include longitudinal matrix yielding and splitting in the unidirectional ply at the crack tip. The shear-lag assumption is used to describe the shear transfer between fibers. The fracture behavior of the laminate is studied as a function of initial crack length, constraint ratio, and width of the debonded zone.

316. Kennedy, J. M.; Fichter, W. B.; and Goree, J. G.: *Opening of an Interface Flaw in a Layered Elastic Half-Plane Under Compressive Loading.* NASA TM-86282, Aug. 1984.

84N32877#

A static analysis is given of the problem of an elastic layer perfectly bonded, except for a frictionless interface crack, to a dissimilar elastic half-plane. The free surface of the layer is loaded by a finite pressure distribution directly over the crack. The problem is formulated using the two dimensional linear elasticity equations. Using Fourier transforms, the governing equations are converted to a pair of coupled singular integral equations. The integral equations are reduced to a set of simultaneous algebraic equations by expanding the unknown functions in a series of Jacobi polynomials and then evaluating the singular Cauchy-type integrals.

317. Nagarkar, A. P.; and Herakovich, C. T.: *Nonlinear Temperature Dependent Failure Analysis of Finite Width Composite Laminates.* NASA CR-162868, Dec. 1979.

80N22409#

A quasi-three dimensional, nonlinear elastic finite element stress analysis of finite width composite laminates including curing stresses is presented. Crossply, angle-ply, and two quasi-isotropic grapite/epoxy laminates are studied. Curing stresses are calculated using temperature dependent elastic properties that are input as percent retention curves, and stresses due to mechanical loading in the form of an axial strain are calculated using tangent moduli obtained by Ramberg-Osgood parameters. It is shown that curing stresses and stresses due to tensile loading are significant as edge effects in all types of laminate studies. The tensor polynomial failure criterion is used to predict the initiation of failure. The mode of failure is predicted by examining individual stress contributions to the tensor polynomial.

318. O'Brien, T. K.: Analysis of Local Delaminations and Their Influence on Composite Laminate Behavior. *Delamination and Debonding of Materials,* ASTM, 1985, pp. 282–297.

86A20639

An equation was derived for the strain energy release rate, G, associated with local delamination growth from a matrix ply crack. The critical rate for edge delamination onset in graphite epoxy laminates was measured and used to predict local delamination onset strains. A simple technique for predicting strain concentrations in the primary load bearing plies near local delaminations was developed. These strain concentrations were responsible for reduced laminate nominal failure strains in laminates containing local delaminations. The influence of edge delamination and matrix crack tip delamination on laminate stiffness and strength was compared.

319. O'Brien, T. K.; Raju, I. S.; and Garber, D. P.: *Residual Thermal and Moisture Influences on the Strain Energy Release Rate Analysis of Edge Delamination.* NASA TM-86437, June 1985.

85N31238#

A laminated plate theory analysis is developed to calculate the strain energy release rate associated with edge delamination growth in a composite laminate. The analysis includes the contribution of residual thermal and moisture stresses to the strain energy released. The strain energy release rate, G, increased when residual thermal effects were combined

with applied mechanical strains, but then decreased when increasing moisture content was included. A quasi-three-dimensional finite element analysis indicated identical trends and demonstrated these same trends for the individual strain energy release rate components.

320. O'Brien, T. K.: *Mixed-Mode Strain-Energy-Release Rate Effects on Edge Delamination of Composites.* NASA TM-84592, Jan. 1983.

83N17598#

Unnotched graphite/epoxy laminates, designed to delaminate at the edges under static and cyclic tensile loads, were tested and analyzed.

321. O'Brien, T. K.: *Characterization of Delamination Onset and Growth in a Composite Laminate.* NASA TM-81940, Jan. 1981.

81N18098#

The onset and growth of delaminations in unnotched graphite epoxy laminates are described quantitatively. These laminates, designed to delaminate at the edges under tensile loads, were tested and analyzed. Delamination growth and stiffness loss were monitored nondestructively. A delamination resistance curve (R curve) was developed to characterize the observed stable delamination growth under quasi-static loading.

322. O'Brien, T. K.; Johnston, N. J.; Morris, D. H.; and Simonds, R. A.: *Determination of Interlaminar Fracture Toughness and Fracture Mode Dependence of Composites Using the Edge Delamination Test.* *Testing, Evaluation and Quality Control of Composites* (Proceedings of the International Conference, Guildford, Surrey, England, Sept. 13–14), Butterworth Scientific, Ltd. (Kent, England), 1983, pp. 223–232.

85A14113

Edge delamination testing has been conducted in order to determine the mixed mode interlaminar fracture toughness of unnotched laminates with graphite fibers and epoxy matrices, with or without elastomer additives. Plots of interlaminar fracture toughness as a function of crack opening energy release rate percentage have been drawn to generate delamination failure criteria which reflect the observed fracture mode dependence.

323. O'Brien, T. K.: *Interlaminar Fracture of Composites.* NASA TM-85768, June 1984.

84N27835#

Fracture mechanics has been found to be a useful tool for understanding composite delamination. Analyses for calculating strain energy release rates associated with delamination growth have been developed. These analyses successfully characterized delamination onset and growth for particular sources of delamination. Low-velocity impact has been found to be the most severe source of composite delamination. A variety of test methods for measuring interlaminar fracture toughness are being developed to identify new composite materials with enhanced delamination resistance.

324. Raju, I. S.; Whitcomb, J. D.; and Goree, J. G.: *A New Look at Numerical Analyses of Free-Edge Stresses in Composite Laminates.* NASA TP-1751, Dec. 1980.

81N13373#

The edge stress problem for a ±45 deg graphite/epoxy laminate was examined. The reliability of the displacement formulated finite element method in analyzing the edge stress problem was investigated. Analyses of two well-known elasticity problems, one involving a stress discontinuity and one a singularity, showed that the finite element analysis yields accurate stress distributions everywhere except in two elements closest to the stress discontinuity or singularity. Stress distributions for a ±45 deg laminate showed the same behavior near the singularity found in the well-known problems with exact solutions. The displacement formulated finite element method appears to be a highly accurate technique for calculating interlaminar stress in composite laminates. The disagreement among the numerical methods was attributed to the unsymmetric stress tensor at the singularity.

325. Raju, I. S.; and Crews, J. H., Jr.: Interlaminar Stress Singularities at a Straight Free Edge in Composite Laminates. *Computers and Structures,* vol. 14, no. 1–2, 1981, pp. 21–28.

82A11229

A quasi-three dimensional finite element analysis was used to analyze the edge stress problem in four-ply composite laminates. The seven laminates that were considered belong to the laminate family where the outer ply angle is between 0 and 90 deg. Systematic convergence studies were made to explore the existence of stress singularities near the free edge. The present analysis appears to confirm the existence of stress singularities at the intersection of the interface and the free edge. The power of the stress singularity was the same for all seven laminates considered.

326. Raju, I. S.; and Crews, J. H., Jr.: *Three-Dimensional Analysis of 0/90s and 90/0s Laminates With a Central Circular Hole.* NASA TM-83300, Apr. 1982.

82N24302#

Stress distributions were calculated near a circular hole in laminates using a three-dimensional-finite element analysis. These stress distributions were presented three ways: through-the-thickness at the hole boundary, along radial lines at the 0/90 and 90/0 interfaces, and around the hole at these interfaces. The interlaminar normal stress and the shear stress distributions had very steep gradients near the hole boundary, suggesting interlaminar stress singularities. The largest compressive stress occurred at about 60 deg from the load axis. A simple procedure was introduced to calculate interlaminar stresses near the hole boundary. It used stresses calculated by an exact two-dimensional analysis of a laminate with a hole as input to a quasi-three-dimensional model. It produced stresses that agreed closely with those from the three-dimensional-finite element model.

327. Ramkumar, R. L.; and Whitcomb, J. D.: Characterization of Mode I and Mixed-Mode Delamination Growth in T300/5208 Graphite/Epoxy. *Delamination and Debonding of Materials*, ASTM, 1985, pp. 315–335.

86A20641

The roles played by Mode I and Mode II strain-energy release rates in inducing delamination growth under static and fatigue loading were investigated, using T300/5208 graphite/epoxy specimens. Double cantilever beam specimens and cracked lap shear specimens were used for pure Mode I and mixed-mode tests, respectively. Fatigue-induced delamination growth was characterized by constant-amplitude fatigue tests at a minimum to maximum cyclic load ratio of 0.05 and a frequency of 10 Hz. During the tests, the maximum and minimum strain-energy release rates and the delamination growth rate were monitored.

328. Shivakumar, K. N.; and Whitcomb, J. D.: *Buckling of a Sublaminate in a Quasi-Isotropic Composite Laminate.* NASA TM-85755, Feb. 1984.

84N18325#

The buckling of an elliptic delamination embedded near the surface of a thick quasi-isotropic laminate was predicted. The thickness of the delaminated ply group (the sublaminate) was assumed to be small compared to the total laminate thickness. Finite-element and Rayleigh-Ritz methods were used for the

analyses. The Rayleigh-Ritz method was found to be simple, inexpensive, and accurate, except for highly anisotropic delaminated regions. Effects of delamination shape and orientation, material anisotropy, and layup on buckling strains were examined.

329. Sohi, M. M.; Hahn, H. T.; and Williams, J. G.: *The Effect of Resin Toughness and Modulus on Compressive Failure Modes of Quasi-Isotropic Graphite/Epoxy Laminates.* NASA TM-87604, Mar. 1986.

86N22647

Compressive failure mechanisms in quasi-isotropic graphite/epoxy laminates were characterized for both unnotched and notched specimens and also following damage by impact. Two types of fibers and four resin systems were studied. For all material combinations, failure of unnotched specimens was initiated by kinking of fibers in the 0-degree plies. A major difference was observed, however, in the mode of failure propagation after the 0-degree ply failure. The strength of quasi-isotropic laminates in general increased with increasing resin tensile modulus. For the materials studied, however, the type of fiber had no effect on the notch sensitivity.

330. Wang, S. S.; and Choi, I.: The Mechanics of Delamination in Fiber-Reinforced Composite Materials. II—The Delamination Behavior and Fracture Mechanics Parameters. *Mechanics of Composite Materials—1983* (Proceedings of the Symposium, Boston, Mass., Nov. 13–18), ASME, 1983, pp. 105–133.

84A22266#

The complete solution structure associated with a composite delamination is determined. Fracture mechanics parameters characterizing the interlaminar crack behavior are defined from asymptotic stress solutions for delaminations with different crack-tip deformation configurations. A numerical method employing singular finite elements is developed to study delaminations in fiber composites with any arbitrary combinations of lamination, material, geometric, and crack variables. The special finite elements include the exact delamination stress singularity in its formulation. The method is shown to be computationally accurate, efficient, and operationally simple.

331. Wang, S. S.; and Choi, I.: The Mechanics of Delamination in Fiber-Reinforced Composite Materials. I—Stress Singularities and Solution Structure. *Mechanics of Composite Materials—1983* (Proceedings of the Symposium, Boston, Mass., Nov. 13–18), ASME, 1983, pp. 83–104.

84A22265#

Stress singularities and complete solution structures associated with general composite delaminations are determined. For a fully open delamination with traction-free surfaces, oscillatory stress singularities always appear, leading to physically inadmissible field solutions. A refined model is introduced by considering a partially closed delamination with crack surfaces in finite-length contact. Stress singularities associated with a partially closed delamination having frictional crack-surface contact are determined and are found to be different from the frictionless-contact case.

332. Webster, J. D.: *Flaw Criticality of Circular Disbond Defects in Compressive Laminates—Graphite-Epoxy Laminates.* NASA CR-164830, June 1981.

81N32195#

The compressive behavior of T300/5208 graphite-epoxy laminates containing circular delaminations was studied to determine the flaw criticality of two types of implanted defects, Kapton bag and Teflon film, on several laminate configurations. Defect size was varied. Results are presented in the form of residual strength curves. The analytically predicted buckling loads show excellent agreement with experimental results and are useful in predicting failure mode transition.

333. Whitcomb, J. D.; and Raju, I. S.: Analysis of Interlaminar Stresses in Thick Composite Laminates With and Without Edge Delamination. *Delamination and Debonding of Materials,* ASTM, 1985, pp. 69–94.

86A20630

The effect of laminate thickness on the interlaminar stresses in rectangular quasi-isotropic laminates under uniform axial strain was studied. Laminates from 8-ply to infinitely thick were analyzed. Thick laminates were synthesized by stacking (45/0/−45/90) ply groups, rather than grouping like plies. Laminates with and without delaminations were studied.

334. Whitcomb, J. D.: Finite-Element Analysis of Instability-Related Delamination Growth. *Journal of Composite Materials,* vol. 15, Sept. 1981, pp. 403–426.

82A21970

Instability-related growth of through-width delaminations in laminated coupons was investigated analytically and experimentally. In the analytical effort, a geometrically nonlinear finite-element stress analysis was used, and the numerical results were compared with exact solutions. In addition, measured lateral deflections of postbuckled through-width delaminations were compared with predicted deflections. Lateral deflections, stress distributions, and strain energy release rates were calculated for various delamination lengths, delamination depths, and applied loads.

335. Whitcomb, J. D.; Raju, I. S.; and Goree, J. G.: Reliability of the Finite Element Method for Calculating Free Edge Stresses in Composite Laminates. *Computers and Structures,* vol. 15, no. 1, 1982, pp. 23–37.

82A18340

The reliability of the displacement-formulated finite element method in analyzing the edge-stress problem of a composite laminate is investigated. The history of the edge-stress problem is reviewed, and two well-known elasticity problems, one involving a stress discontinuity and one a singularity, are analyzed. The finite element analysis in these problems yields accurate stress distributions everywhere except in two elements closest to the stress discontinuity or singularity. Stress distributions for a ±45 deg ply laminate near the singularity were similar to those of the two elasticity problems.

336. Whitcomb, J. D.: Strain-Energy Release Rate Analysis of Cyclic Delamination Growth in Compressively Loaded Laminates. *Effects of Defects in Composite Materials,* ASTM, 1984, pp. 175–193.

85A30510

Delamination growth in compressively loaded composite laminates was studied analytically and experimentally. The configuration used was a laminate with an across-the-width delamination. An approximate super-position stress analysis was developed to quantify the effects of various geometric, material, and load parameters on strain-energy release rates.

337. Whitcomb, J. D.; and Raju, I. S.: Super-position Method for Analysis of Free-Edge Stresses. *Journal of Composite Materials,* vol. 17, Nov. 1983, pp. 492–507.

84A25002

Superposition techniques were used to transform the edge stress problem for composite laminates into a more lucid form. By eliminating loads and stresses not contributing to interlaminar stresses, the essential aspects of the edge stress problem are easily recognized. Transformed problem statements were developed for both mechanical and thermal loads. Also, a technique for approximate analysis using a two-dimensional plane strain analysis was developed.

338. Whitcomb, J. D.: *Finite Element Analysis of Instability-Related Delamination Growth.* NASA TM-81964, Mar. 1981.

81N21132#

A parametric study of postbuckled through-width delaminations in laminated coupons was performed. A finite element analysis was developed to analyze the coupons as a combination of linear and geometrically nonlinear components. Because most of the coupon configurations studied behave linearly, the mixed linear and nonlinear analysis greatly reduced computational costs. The analysis was verified by comparing numerical with exact solutions for simple hypothetical problems. In addition, measured lateral deflections of postbuckled through-width delaminations in laminated coupons were compared with predicted deflections.

339. Whitcomb, J. D.: *Approximate Analysis of Postbuckled Through-Width Delaminations.* NASA TM-83147, June 1981.

81N27202#

An approximate analysis was developed to analyze the postbuckling behavior of through-width delaminations in a laminated coupon. The analysis contains two parameters which are determined using a finite element analysis. After calculating the parameters for a few configurations, the approximate analysis was used to analyze many other configurations. Lateral deflections and mode I strain-energy release rates obtained with the approximate analysis were compared with results from the finite element analysis. For the configurations analyzed, the approximate analysis agreed very well with the finite element results.

340. Whitcomb, J. D.: *Analysis of Instability-Related Growth of a Through-Width Delamination.* NASA TM-86301, Sept. 1984.

84N33524#

The effect of various parameters on instability-related delamination growth was studied. The configuration studied consisted of a thick composite laminate with a single through-width delamination located near one surface. Both mechanical and thermal loads were considered.

341. Williams, J. H., Jr.: *Dynamic Fracture of Advance Fiber Composites.* NASA CR-155338, Dec. 1977.

78N13135#

Efforts to investigate some of the theoretical bases for the dynamic fracture of graphite fiber reinforced composites are described. In particular, the initiation of unstable fracture in unidirectional cracked composites, dynamic crack propagation as modeled by orthotropic double cantilever beams, and interlaminar stresses for laminates having arbitrary stacking sequences are summarized.

Environmental effects and flight service

342. Baker, D. J.: *Flight Service Evaluation of Composite Components on Bell 206L and Sikorsky S-76 Helicopters.* NASA TM-84637, Mar. 1983.

83N19815#

Progress on two programs to evaluate composite structural components in flight service on commercial helicopters is described. Thirty-six ship sets of composite components that include the litter door, baggage door, forward fairing, and vertical fin were installed on Bell Model 206L helicopters that are operating in widely different climatic areas. Results achieved from 14,000 hours of accumulated service on the 206L components, tests on an S-76 horizontal stabilizer after 1600 hours of service, tests on an S-76 tail rotor spar after 2300 hours service, and two years of ground based exposure of material coupons are reported.

343. Baker, D. J.: *Ground Exposure of Composite Materials for Helicopters.* NASA TM-86311, Oct. 1984.

84N34577#

Residual strength results are presented on four composite material systems that were exposed for three years at locations on the North American Continent. The composite systems are: (1) Kelvar-49 fabric/F-185 epoxy; (2) Kevlar-49 fabric/LRF-277 epoxy; (3) Kevlar-49 fabric/CE-306 epoxy; and (4) T-300 Graphite/E-788 epoxy.

344. Chapman, A. J.; Hoffman, D. J.; and Hodges, W. T.: Effects of Commercial Aircraft Operating Environment on Composite Materials. *The 1980's— Payoff Decade for Advanced Materials* (Proceedings of the Twenty-Fifth National Symposium and Exhibition, San Diego, Calif., May 6–8), Society for the Advancement of Material and Process Engineering, 1980, pp. 737–751.

81A22681

Composite specimens configured for various mechanical property tests are exposed to environmental conditions on aircraft in scheduled airline service, on racks at major airports, and to controlled environmental conditions in the laboratory. Results of tests following these exposures will identify critical parameters affecting composite durability, and correlation of the data will aid in developing methods for predicting durability. Interim results of these studies show that mass change of composite specimens on commercial aircraft depends upon the regional climate and season, and that mass loss from composite surfaces due to ultraviolet radiation can be largely prevented by aircraft paint.

345. Coggeshall, R. L.: *Environmental Exposure Effects on Composite Materials for Commercial Aircraft.* NASA CR-177929, Nov. 1985.

86N18449#

The effects of environmental exposure on composite materials are determined. The environments considered are representative of those experienced by commercial jet aircraft. Initial results have been compiled for the following material systems: T300/5208, T300/5209, and T300/934. Residual strength results are presented for specimens exposed for up to five years at five ground-level exposure locations and on airplanes from one airline.

346. Coggeshall, R. L.: *The 737 Graphite Composite Flight Spoiler Flight Service Evaluation.* NASA CR-172600, July 1985.

86N18448#

The flight service experience of 111 graphite-epoxy spoilers on 737 transport aircraft and related ground based environmental exposure of graphite-epoxy material specimens are covered. Spoilers have been installed on 28 aircraft representing seven major airlines operating throughout the world. An extended flight service evaluation program of 15 years is presently underway. As of December 1984, a total of 2,092,155 spoiler flight hours and 2,954,814 spoiler landings had been accumulated by this fleet.

347. Cohen, D.; Hyer, M. W.; and Tompkins, S. S.: The Effects of Thermal Cycling on Matrix Cracking and Stiffness Changes in Composite Tubes. *Proceedings of the 16th National Technical Conference* (Albuquerque, N. Mex., Oct. 9–11), Society for the Advancement of Material and Process Engineering, 1984, pp. 577–588.

85A37399

The study investigated the accumulation of transverse matrix cracks and the resultant loss of torsional, extensional, and bending stiffnesses in 8 layer 0.5 in. diameter crossply tubes subjected to thermal cycling. The tubes were graphite-epoxy and the temperature range during cycling was −250 to 200°F. The effect of fiber and matrix properties was investigated through the use of T300 and P75S fibers and 934 and CE339 resins. The study considered 0, 10, 50, 100, 300, and 500 thermal cycles.

348. Dexter, H. B.: *Composite Components on Commercial Aircraft.* NASA TM-80231, Mar. 1980.

80N18109#

Flight service programs with advanced composites sponsored by NASA during the past 10 years are described. Approximately 2.5 million total composite component flight hours have been accumulated since 1970 on both commercial transports and helicopters. Design concepts with significant mass savings were developed, appropriate inspection and maintenance procedures were established, and satisfactory service was achieved for the various composite components. Ground and flight environmental effects on the composite materials used in the flight service programs supplement the flight service evaluation.

349. Dexter, H. B.; and Chapman, A. J.: NASA Service Experience With Composite Components. *Materials 1980* (Proceedings of the Twelfth National Technical Conference, Seattle, Wash., Oct. 7–9), Society for the Advancement of Material and Process Engineering, 1980, pp. 77–99.

81A43607

Flight service experience is reported for more than 140 composite aircraft components with up to 8 years service and almost two million successful component flight hours. Composite components are being evaluated on Boeing, Douglas, and Lockheed transport aircraft. Components are currently under development for service evaluation on Bell and Sikorsky helicopters. Design concepts and inspection and maintenance results are reported for com-

ponents currently in service. Results of flight, outdoor ground, and controlled laboratory environmental tests on composite materials used in the flight service programs are also presented.

350. Dexter, H. B.: Composite Components on Commercial Aircraft. Paper presented at the AGARD Specialists Meeting on the Effect of Service Environment on Composite Materials, Athens, Greece, Apr. 13–18, 1980.

80A27597#

The paper considers the use of composite components in commercial aircraft. NASA has been active in sponsoring flight service programs with advanced composites for the last 10 years, with 2.5 million total composite component hours accumulated since 1970 on commercial transports and helicopters with no significant degradation in residual strength of composite components. Design, inspection, and maintenance procedures have been developed.

351. Gibbins, M. N.; and Hoffman, D. J.: *Environmental Exposure Effects on Composite Materials for Commercial Aircraft.* NASA CR-3502, Jan. 1982.

82N16178#

The effects of environmental exposure on composite materials are studied. The environments considered are representative of those experienced by commercial jet aircraft. Results have been compiled for the following material systems: T300/5208, T300/5209, and T300/934. Specimens were exposed on the exterior and interior of Boeing 737 airplanes of three airlines, and to continuous ground level exposure at four locations. In addition specimens were exposed in the laboratory to conditions such as simulated ground-air-ground, weatherometer, and moisture.

352. Hendricks, C. L.: *A Study of the Effects of Long-Term Exposure to Fuels and Fluids on the Behavior of Advanced Composite Materials.* NASA CR-158169, Sept. 1977.

79N17918#

The Task 1 (thirty-six-month) and Task 2 (twelve-month) exposures of composite materials to fuel and fluid environments are reported. Narmco T300/5209 (Task 1) shows no significant degradation. Slightly lower mechanical properties were obtained from fuel/water immersion exposure of this material. Kevlar 49/2544 (also Task 1) exhibits a significant drop in short-beam shear when exposed to fuel/water immersion. Task 2 materials (T300/5208, Kevlar 49 fabric/5209, and Kevlar 49/fibers) have not shown any significant mechanical property degradation.

353. Hoffman, D. J.: *Environmental Exposure Effects on Composite Materials for Commercial Aircraft.* NASA CR-165649, Aug. 1980.

81N16139#

The test program concentrates on three major areas: flight exposure, ground based exposure, and accelerated environmental effects and data correlation. Among the parameters investigated were: geographic location, flight profiles, solar heating effects, ultraviolet degradation, retrieval times, and test temperatures. Data from the tests can be used to effectively plan the cost of production and viable alternatives in materials selection.

354. Hoffman, D. J.; and Stoecklin, R. L.: *The 737 Graphite Composite Flight Spoiler Flight Service Evaluation.* NASA CR-159094, Jan. 1980.

80N17147#

The flight service experience of 111 graphite-epoxy spoilers on 737 transport aircraft and related ground-based environmental exposure of graphite-epoxy material specimens is reported. Spoilers were installed on 28 aircraft representing seven major airlines operating throughout the world. Over 1,188,367 spoiler flight hours and 1,786,837 spoiler landings were accumulated by this fleet. Tests of removed spoilers and ground-based exposure specimens after the fifth year of service indicate modest changes in composite strength properties. Two incidents of trailing edge delamination with subsequent core corrosion were observed. Based on visual, ultrasonic, and destructive testing, there has been no evidence of moisture migration into the honeycomb core and no core corrosion.

355. Howell, W. E.; and Plumer, J. A.: *Effects of Simulated Lightning on Composite and Metallic Joints.* NASA TM-84554, Oct. 1982.

83N14170#

The effects of simulated lightning strikes and currents on aircraft bonded joints and access/inspection panels were investigated. Both metallic and composite specimens were tested. Tests on metal-fuel-feed-through-elbows in graphite/epoxy structures were evaluated. Sparking threshold and residual strength of single lap bonded joints and sparking threshold of access/inspection panels and metal-fuel-feed-through elbows are reported.

356. Lehman, G. H.: *Flight-Service for Advanced Composite Rudders on Transport Aircraft.* NASA CR-174537, July 1979.

85N29950

Flight service experience and in-service inspection results are reported for DC-10 graphite composite rudders during the third year of airline service.

357. Long, E. R., Jr.: *Moisture Diffusion Parameter Characteristics for Epoxy Composites and Neat Resins.* NASA TP-1474, Aug. 1979.

79N29241

The moisture absorption characteristics of two graphite/epoxy composites and their corresponding cured neat resins were studied in high humidity and water immersion environments at elevated temperatures. Moisture absorption parameters such as equilibrium moisture content and diffusion coefficient, derived from data taken on samples exposed to high humidity and water soak environments, were compared. Composite swelling in a water immersion environment was measured. Tensile strengths of cured neat resin were measured as a function of their equilibrium moisture content after exposure to different moisture environments. The effects of intermittent moderate tensile loads on the moisture absorption parameters of composite and cured neat resin samples were determined.

358. Pater, R. H.; and Scola, D. A.: Isothermal and Moisture Effects on Cloth Reinforced Resin Composites. *New Horizons—Materials and Processes for the Eighties* (Proceedings of the Eleventh National Conference, Boston, Mass., Nov. 13–15), Society for the Advancement of Material and Process Engineering, 1979, pp. 645–662.

80A34795

A series of cloth reinforced/resin composites were evaluated with a view of selecting commercially available materials which have potential for ease of processing, and which represent three upper temperature limits of 149°C, 204°C, and 288°C for long-term use. Twelve prepreg materials obtained from a commercial source were evaluated. The composite systems were subjected to isothermal aging at 178°C in air for time periods up to 600 hours. The effects of these isothermal aging conditions on the room temperature flexural strength and modulus are presented. The moisture absorption behavior after exposure to boiling water for 13 days and the effects of boiling water exposure on the flexural properties of these composite materials after 24 and 112 hours are also discussed.

359. Pittman, C. M.; and Brown, R. D.: *Exploratory Investigation of Two Resin-Matrix Composites Subjected to Arc-Tunnel Heating.* NASA TP-1429, May 1979.

79N23141#

Flexure and short beam shear specimens of a graphite epoxy and a graphite polyimide composite material were tested in an arc-tunnel environment. The materials were subjected to seven heating rates from 12.5 to 308 kW/sw m for test times from 5 to 1000 seconds. The temperature response of the specimens was measured and the strength degradation caused by the various heating environments was determined.

360. Pride, R. A.: Interim Results of Long-Term Environmental Exposures of Advanced Composites for Aircraft Applications. *Proceedings of the 11th Congress of the International Council of the Aeronautical Sciences* (Lisbon, Portugal, Sept. 10–16), *Volume 1*, International Council of the Aeronautical Sciences Secretariat (DGLR), 1978, pp. 234–241.

79A20096#

Interim results from a number of ongoing, long-term environmental effects programs for composite materials are reported. The flight service experience is evaluated for 142 composite aircraft components after more than five years and one million successful component flight hours. Ground-based outdoor exposures of composite material coupons after 3 years of exposure at five sites have reached equilibrium levels of moisture pickup which are predictable. Solar ultraviolet-induced material loss is discussed for these same exposures. No significant degradation has been observed in residual strength for either stressed or unstressed specimens, or for exposures to aviation fuels and fluids.

361. Rasch, N. O.: *A Compendium of Lightning Effects on Future Aircraft Electronic Systems* (Proceedings of Conference, Hampton, Va., Nov. 4–6, 1981). DOT/FAA/CT-82/30, Feb. 1982.

82N28293#

This publication is a composite of presentations given at the NASA Langley Research Center/FAA Technical Center "Lightning Effects on Future Aircraft Systems Workshop" held on November 4–6, 1981, at the NASA Langley Research Center. The presentations encompassed the full spectrum of lightning research from lightning phenomenology, lightning modeling, electromagnetic issues associated with composite materials, to the lightning/aircraft electromagnetic interaction analysis. Also included are a total of five presentations assessing the Digital System upset phenomena.

362. Stoecklin, R. L.: *Development, Manufacturing, and Test of Graphite-Epoxy Composite Spoilers for Flight Service on 737 Transport Aircraft.* NASA CR-132682, Oct. 1976.

77N16023#

A total of 114 spoiler units were fabricated in a production shop environment, utilizing three graphite-epoxy material systems. The graphite-epoxy skins were laid up on production tooling using both mechanical and hand layup techniques. Inspection techniques utilized MRB type assessment in the abscence of quality requirements. Each completed spoiler was subjected to ultrasonic inspection utilizing a multicolor recording system that documented each inspection result. In addition, one static test spoiler was sectioned after the test to examine the adhesive filleting to the honeycomb core. Visual examination of the cured adhesives showed excellent results.

363. Stoecklin, R. L.: *The 737 Graphite Composite Flight Spoiler Flight Service Evaluation.* NASA CR-145207, Aug. 1977.

78N10047#

The flight service experience of 110 graphite epoxy spoilers on 737 transport aircraft was reviewed. Several spoilers were installed on each of 27 aircraft representing seven major airlines operating throughout the world. A flight service evaluation program of at least 5 years is under way. As of April 30, 1977, a total of 766,938 spoiler flight hours and 1,168,090 spoiler landings were accumulated by the fleet. Based on visual ultrasonic and destructive testing, there was no evidence of moisture migration into the honeycomb core and no core corrosion.

364. Stone, R. H.: *Flight Service Evaluation of Advanced Composite Ailerons on the L-1011 Transport Aircraft.* NASA CR-177960, July 1985.

86N14317#

Flight service evaluation of composite inboard ailerons on the L-1011 under Contract NAS1-15069 for a period of five years is discussed. No visible damage was observed on any of the composite ailerons, and no maintenance action has occurred on any of the composite parts except for repainting of areas with paint loss. Flight hours on the airline components at the time of inspection ranged from 8,787 to 10,804 hours, after approximately 3 years of service.

365. Stone, R. H.: *Flight Service Evaluation of Kevlar-49 Epoxy Composite Panels in Wide-Bodied Commercial Transport Aircraft.* NASA CR-172344, June 1984.

84N31287

Kevlar-49 fairing panels, installed as flight service components on three L-1011s, were inspected after 10 years of service. The Kevlar-49 components were found to be performing satisfactorily in service with no major problems or any condition requiring corrective action. The only defects noted were minor impact damage, a few minor disbonds, and a minor degree of fastener hole fraying and elongation. These defects are for the most part comparable to damage noted on fiberglass fairings. The service history obtained in this program indicates that Kevlar-49 epoxy composite materials have satisfactory service characteristics for use in aircraft secondary structures.

366. Sykes, G. F.; Burks, H. D.; and Nelson, J. B.: The Effect of Moisture on the Dynamic Thermomechanical Properties of a Graphite/Epoxy Composite. *Diversity—Technology Explosion* (Proceedings of the Twenty-Second National Symposium and Exhibition, San Diego, Calif., Apr. 26–28), Society for the Advancement of Material and Process Engineering, 1977, pp. 350–364.

78A16893

A study has been made of the effect of moisture absorption on the dynamic thermomechanical properties of a graphite/epoxy composite recently considered for building primary aircraft structures. Torsional braid analysis and thermomechanical analysis techniques were used to measure changes in the glass transition temperature and the initial softening temperature of T-300/5208 graphite/epoxy composites exposed to room temperature water soak.

367. Tanimoto, E. Y.: *A Study of the Effects of Long-Term Exposure to Fuels and Fluids on the Behavior of Advanced Composite Materials.* NASA CR-165763, Aug. 1981.

82N18327#

The periodic testing and evaluation of graphite/epoxy and Kevlar/epoxy material systems, after subjecting test specimens to prolonged exposure to several laboratory-controlled environments deemed typical of normal aircraft operations, are discussed. It is noted that specimen immersion in water or water-based fluids resulted in the greatest effect on the mechanical properties tested. Also, the environmental fluids showed a tendency to affect Kevlar/epoxy

systems at an earlier exposure period than the graphite/epoxy systems. Results also indicate mechanical property strength retention generally being lower for the Kevlar/epoxy systems when compared to the corresponding graphite/epoxy systems in similar environments, after prolonged exposure.

368. Tompkins, S. S.: *Analytical Study of Effects of Surface and Environmental Thermal Properties on Moisture in Composites.* NASA TM X-3562, Sept. 1977.

77N32247#

The influence of surface and environmental thermal properties on the moisture absorption in fiber-reinforced polymeric-resin-matrix composite materials which have been subjected to convection and solar radiation was studied. Predicted moisture contents based on the conditions at the heated surface and the ambient air were compared for both short term and long term exposures over a wide range of values for emittance, solar absorptance, convective heat transfer coefficient, solar radiation, ambient temperature, and orientation of the surface with respect to the sun. The calculations showed that absorptance and heat transfer coeffficients have significant effect on the moisture content.

369. Tompkins, S. S.; Tenney, D. R.; and Unnam, J.: *Prediction of Moisture and Temperature Changes in Composites During Atmospheric Exposure.* NASA TM-78711, Apr. 1978.

78N23149#

The effects of variations in diffusion coefficients, surface properties of the composite, panel tilt, ground reflection, and geographical location on the moisture concentration profiles and average moisture content of composite laminates were studied analytically. A heat balance which included heat input due to direct and sky diffuse solar radiation, ground reflection, and heat loss due to reradiation and convection was used to determine the temperature of composites during atmospheric exposure.

370. Unnam, J.; and Tenney, D. R.: *Analytical Prediction of Moisture Absorption/Desorption in Resin Matrix Composites Exposed to Aircraft Environments. 18th AIAA Structures, Structural Dynamics and Materials Conference and Aircraft Composites: The Emerging Methodology for Structural Assurance* (San Diego, Calif., Mar. 21–25), *Technical Papers, Volume A*, 1977, pp. 227–235. (Available as AIAA Paper 77-400.)

77A25752#

The moisture absorption/desorption behavior of resin matrix composites was mathematically modeled by classical diffusion theory using an effective diffusion coefficient. Good agreement was found between calculated moisture content and published data for T300-5208 graphite fiber reinforced epoxy matrix composite. Weather Bureau data for Langley Air Force Base and Norfolk, Va., were used to calculate the amount of moisture that a T300/5208 composite panel would contain if exposed outdoors. Results obtained by using average monthly weather data for several high aircraft usage locations around the world suggest that, except for desert areas, geographical locations should have only minimal effect on the moisture absorption level reached in composites. Solar radiation data together with cloud and wind information were included in the analysis to estimate an effective temperature of the composite panel during ground exposure.

Materials development and processing

371. Biermann, T. F.; and Hopper, L. C.: *Development of a Heterogeneous Laminating Resin System.* NASA CR-172459, Jan. 1985.

85N18089#

The factors which affect the impact resistance of laminating resin systems and yet retain equivalent performance with the conventional 450 K curing epoxy matrix systems in other areas were studied. Formulation work was conducted on two systems, an all-epoxy and an epoxy/bismaleimide, to gain fundamental information on the effects formulation changes have upon neat resin and composite properties.

372. Bluck, R. M.; and Johnson, R. R.: New Innovations in Composites. *Technology Vectors* (Proceedings of the Twenty-Ninth National SAMPE Symposium and Exhibition, Reno, Nev., Apr. 3–5), Society for the Advancement of Material and Process Engineering, 1984, pp. 1115–1126.

85A16046

Procedures for manufacturing metal clad graphite resin structural elements applicable to optical systems requiring precisely controlled thermal and mechanical properties are described.

373. Bonnar, G. R.; and Palmer, R. J.: *Woven Graphite Epoxy Composite Test Specimens With Glass Buffer Strips.* NASA CR-165871, May 1982.

82N23268#

Woven unidirectional graphite cloth with bands of fiberglass replacing the graphite in discrete lengthwise locations was impregnated with epoxy resin and used to fabricate a series of composite tensile and shear specimens. The finished panels, with the fiberglass buffer strips, were tested. Details of the fabrication process are reported.

374. Bradley, R.: *Pultrusion of Graphite/Epoxy Prepreg for Flat and Hat Section Specimens.* NASA CR-158899, Mar. 1978.

78X10122#

Flat and hat shaped specimens were pultruded from graphite/epoxy prepreg using equipment which cures the part simultaneous with continuous passage through a pneumatically actuated ceramic compaction and shaping die. Room temperature mechanical tests were performed on material trimmed from pultruded parts and the data were compared to tests of flat autoclave cured layups. Costs for future production runs of pultruded flat and hat shaped structural elements were estimated.

375. Brand, R. A.; and Harrison, E. S.: *Development of Tough, Moisture Resistant Laminating Resins.* NASA CR-3615, Sept. 1982.

82N33432#

Tough, moisture resistant laminating resins for employment with graphite fibers were developed. The new laminating resins exhibited cost, handleability, and processing characteristics equivalent to 394 K (250°F) curing epoxies. The laminating resins were based on bisphenol A dicyanate and monofunctional cyanates with hydrophobic substituents. Toughening was accomplished by the precipitation of small diameter particles of butadiene nitrile rubber throughout the resin matrix.

376. Brand, R. A.; and Harrison, E. S.: *Feasibility Study on the Development of Tough, Moisture-Resistant Laminating Resins.* NASA CR-3185, Oct. 1979.

79N33259#

The potential of cyanate resins as replacements for epoxy resins in composites with graphite fiber reinforcement was investigated in an effort to provide improved moisture resistance and toughness in laminating systems at a projected cost, handleability, and processing requirements equivalent to 400 K (260°F) curing epoxies. Monomer synthesis, formulation, blending, resin preparation, catalysis studies, prepreg preparation, laminate fabrication, and testing are discussed.

377. Chung, H. H.: *Study of Low-Cost Fabrication Methods for Aerospace Composite Materials.* NASA CR-145330, Jan. 1978.

78N19205#

The wet pultrusion process shows promise of being a low-cost method for producing composite parts with constant cross section along the length. A cost analysis showed at least 80 percent cost reduction for the hat section and 40 percent for the flat panel by pultrusion over the conventional manual and automated layups.

378. Freeman, W. T., Jr.; and Stein, B. A.: *Filament Winding—Waking the Sleeping Giant. Aerospace America,* vol. 24, Oct. 1985, pp. 44–46, 48, 49.

86A10146#

The use of filament winding (FW) in the production of aerospace composite structures is examined. The FW process applies spools of fiber and prepreg tow or prepreg tape to a male mandrel; the process is more efficient and cost effective than metallic construction. The fibers used in FW and the curing process are explained. The reduced storage and fabrication costs that result from FW are discussed. The use of FW to produce a filament-wound case for a solid rocket motor and the substructure and skin of an aircraft fuselage is described. Areas which require further development in order to expand the use of FW are listed and discussed.

379. Gosnell, R. B.: *Feasibility Study on the Development of a Heterogeneous Matrix for Composites.* NASA CR-159204, Apr. 1980.

80X10145#

The development of a toughened resin for use with fibrous reinforcement suitable for commercial aircraft components is discussed with emphasis on the modification of an epoxy with an elastomer. It was determined that carboxy-terminated butadiene-acrylonitrile rubbers are not suitable for modifying Araldite MY720. An alternate liquid elastomer, an amine-terminated butadiene-acrylonitrile rubber, was found to be compatible with the same resin. Only marginal improvement in toughness was observed.

380. Hergenrother, P. M.; Jensen, B. J.; and Havens, S. J.: Thermoplastic Composite Matrices With Improved Solvent Resistance. *SAMPE Journal,* vol. 20, Sept.–Oct. 1984, pp. 18–23.

84A49347

In order to improve solvent resistance of aromatic thermoplastic polymers, ethynyl-terminated aromatic sulfone polymers (ETS), sulfone/ester polymers (SEPE) containing pendent ethynyl groups, and phenoxy resin containing pendent ethynyl groups were synthesized. Cured polysulfones and phenoxy resins containing ethynyl groups on the ends or pendent on the molecules exhibited systematic behavior in solvent resistance, film flexibility, and toughness as a function of crosslink density. The film and composite properties of a cured solvent-resistant ETS were better than those of a commercially available solvent sensitive polysulfone. The study was part of a NASA program to better understand the trade-offs between solvent resistance, processability, and mechanical properties which may be useful in designing composite structures for aerospace vehicles.

381. Hodges, W. T.; and St. Clair, T. L.: Ultrasonic Mixing of Epoxy Curing Agents. *SAMPE Quarterly*, vol. 14, Apr. 1983. pp. 46–50.

84A14290

A new technique for mixing solid curing agents into liquid epoxy resins using ultrasonic energy was developed. This procedure allows standard curing agents to be mixed without prior melting of the curing agent. It also allows curing agents with very high melt temperatures to be mixed without prior melting of the curing agent. It also allows curing agents with very high melt temperatures to be mixed without premature curing.

382. Hodges, W. T.; Tyeryar, J. R.; and Berry, M.: Bonding and Nondestructive Evaluation of Graphite/PEEK Composite and Titanium Adherends With Thermoplastic Adhesives. SME Paper MF85-511, 1985.

86A24669#

Bonded single overlap shear specimens were fabricated from Graphite/PEEK (Polyetheretherketone) composite adherends and titanium adherends. Six advanced thermoplastic adhesives were used for the bonding. The specimens were bonded by an electromagnetic induction technique producing high heating rates and high-strength bonds in a few minutes. Nondestructive evaluation of bonded specimens was performed ultrasonically by energizing the entire thickness of the material through the bondline and measuring acoustic impedance parameters. Destructive testing confirmed the unique ultrasonic profiles of strong and weak bonds, establishing a standard for predicting relative bond strength in subsequent specimens.

383. Hou, T. H.: *Autoclave Processing for Composite Material Fabrication. 1: An Analysis of Resin Flows and Fiber Compactions for Thin Laminate.* NASA CR-178011, Nov. 1985.

86N16270#

A newly formulated resin flow model for composite prepreg lamination process is reported. This model considers viscous resin flows in both directions perpendicular and parallel to the composite plane. In the horizontal direction, a squeezing flow between two nonporous parallel plates is analyzed, while in the vertical direction, a poiseuille type pressure flow through porous media is assumed. Proper force and mass balances have been made and solved for the whole system. The effects of fiber-fiber interactions during lamination are included as well. The unique features of this analysis are: (1) the pressure gradient inside the laminate is assumed to be generated from squeezing action between two adjacent approaching fiber layers, and (2) the behavior of fiber bundles is simulated by a Finitely Extendable Nonlinear Elastic (FENE) spring.

384. Ketterer, M. E.: *Thermoplastic-Carbon Fiber Hybrid Yarn.* NASA CR-3849, Nov. 1984.

85N12093#

Efforts were directed to develop processing methods to make carbon fiber/thermoplastic fiber preforms that are easy to handle and drapeable, and to consolidate them into low-void content laminates. The objectives were attained with the development of the hybrid yarn concept, whereby thermoplastic fiber can be intimately intermixed with carbon fiber into a hybrid yarn. This was demonstrated with the intermixing of Celion 3000 with a Celanese liquid crystal polymer fiber, polybutylene terephthalate fiber, or polyetheretherketone fiber. The intermixing of the thermoplastic matrix fiber and the reinforcing carbon fiber give a preform that can be easily fabricated into laminates with low-void content. Mechanical properties of the laminates were not optimized; however, initial results indicated properties typical of a thermoplastic/carbon fiber composite prepared by more conventional methods.

385. Kranbuehl, D.; Delos, S.; Yi, E.; Mayer, J.; and Hou, T.: Correlation of Dynamic Dielectric Measurements With Viscosity in Polymeric Resin Systems. *Proceedings of the 30th National SAMPE Symposium and Exhibition* (Anaheim, Calif., Mar. 19–21), Society for the Advancement of Material and Process Engineering, 1985, pp. 638–648.

86A13115

Dynamic dielectric analysis has been used to study curing polymer systems and thermoplastics. Measurements have been made over a frequency range of six decades. This wide range of frequencies increases the amount of information which can be obtained. The data are analyzed in terms of the frequency dependence of the specific conductivity and the relaxation time, parameters which are characteristic of the cure state of the material and independent of the size of the sample.

386. Kranbuehl, D.: *Molecular Weight: Property Relationships of High Performance Polymers Used for Adhesives and Composites.* NASA CR-143511, Aug. 1975.

75N32247#

Degradation of high performance polyimide precursor resins was investigated by measuring the molecular weight of the polymers in solution, using a membrane osmometer. It was found that polyimide precursor resins composed of BTDA and ODPA combined with DABP and MDA were unstable in DMAC. The degradation rate was found to depend upon the chemical nature of the isomeric diamine and the geometric structure about the amide linkage.

387. Kray, R. J.: *Development of New Addition-Type Composite Resins.* NASA CR-3276, Apr. 1980.

81N25147#

The most promising of a number of new addition-type polyimides and polyaromatic melamine (NCNS) resins were examined for use in high performance composite materials.

388. Smith, A. C.: Applications of Mass Spectrometry Techniques to Autoclave Curing of Materials. Paper presented at the U.S. Department of Energy, USAF, U.S. Navy, and DARPA Review of Progress in Quantitative NDE, University of California, Santa Cruz, Calif., Aug. 7–12, 1983.

83A50143#

In this paper data are presented for two materials to support the feasibility of using mass spectrometer gas analysis techniques to enhance the knowledge of autoclave curing of composite materials and provide additional information for process control evaluation. It is expected that this technique will also be useful in working out the details involved in determining the proper cure cycle for new or experimental materials.

389. Stein, B. A.; Tyeryar, J. R.; and Hodges, W. T.: *Rapid Adhesive Bonding Concepts.* NASA TM-86256, June 1984.

84N29968#

Adhesive bonding in the aerospace industry typically utilizes autoclaves or presses which have considerable thermal mass. As a consequence, the rates of heatup and cooldown of the bonded parts are limited and the total time and cost of the bonding process is often relatively high. Rapid adhesive bonding concepts were developed to utilize induction heating techniques to provide heat directly to the bond line and/or adherends without heating the entire structure, supports, and fixtures of a bonding assembly. Bonding times for specimens are cut by a factor of 10 to 100 compared to standard press bonding. The development of rapid adhesive bonding for lap shear specimens (per ASTM D1003 and D3163), for aerospace panel bonding, and for field repair needs of metallic and advanced fiber reinforced polymeric matrix composite structures is reviewed.

390. Stein, B. A.; Tyeryar, J. R.; and Hodges, W. T.: Rapid Adhesive Bonding of Advanced Composites and Titanium. AIAA Paper 85-0750, 1985.

85A30406#

Rapid adhesive bonding (RAB) concepts utilize a toroid induction technique to heat the adhesive bond line directly. This technique was used to bond titanium overlap shear specimens with 3 advanced thermoplastic adhesives and APC-2 (graphite/PEEK) composites with PEEK film. Bond strengths equivalent to standard heated-platen press bonds were produced with large reductions in process time. RAB produced very strong bonds in APC-2 adherend specimens; the APC-2 adherends were highly resistant to delamination. Thermal cycling did not significantly affect the shear strengths of RAB titanium bonds with polyimide adhesives. A simple ultrasonic nondestructive evaluation process was found promising for evaluating bond quality.

391. Stinchcomb, W. W.; Henneke, E. G.; and Price H. L.: Use of the Short-Beam Shear Test for Quality Control of Graphite-Polyimide Laminates. *Reproducibility and Accuracy of Mechanical Tests*, ASTM Special Technical Publication No. 626, 1977, pp. 96–109.

77A47966

The short-beam shear test is used widely by both manufacturers and researchers as a quality control

test in the production of materials and development of new material systems. There are, however, several limitations to the standard test method. This paper presents the results of short-beam shear tests on graphite-polyimide laminates and reports on stiffness-strength relationships and nondestructive evaluation methods which aid in the interpretation of the test data.

392. Stoakley, D. M.; and St. Clair, A. K.: *Metal Ion-Containing Epoxies.* NASA TM-84567, Oct. 1982.

83N14272

A variety of metallic and organometallic complexes to be used as potential additives for an epoxy used by the aerospace industry as a composite matrix resin were investigated. A total of 9 complexes were screened for compatibility and for their ability to accelerate or inhibit the cure of a highly crosslinkable epoxy resin. Methods for combining the metallic complexes with the resin were investigated, gel times recorded, and cure exotherms studied by differential scanning calorimetry. Glass transition temperatures of cured metal ion containing epoxy castings were determined by thermomechanical analysis. Thermal stabilities of the castings were determined by thermogravimetric analysis. Mechanical strength and stiffness of these doped epoxies were also measured.

393. Swann, R. T.; and Esterling, D. M.: Chemical Feasibility of Lithium as a Matrix for Structural Composites. *Composites*, vol. 15, Oct. 1984, pp. 305–309.

85A15623

The chemical compatibility of lithium with tows of carbon and aramid fibers and silicon carbide and boron monofilaments was investigated by encapsulating the fibers in liquid lithium and also by sintering. The lithium did not readily wet the various fibers. In particular, very little lithium infiltration into the carbon and aramid tows was achieved and the strength of the tows was seriously degraded. The strength of the boron and silicon carbide monofilaments, however, was not affected by the liquid lithium. Therefore lithium is not feasible as a matrix for carbon and aramid fibers, but a composite containing boron or silicon carbide fibers in a lithium matrix may be feasible for specialized applications.

394. Young, P. R.; and Chang, A. C.: Prepreg Cure Monitoring Using Diffuse Reflectance—FTIR. *Proceedings of the 16th National Technical Conference* (Albuquerque, N. Mex., Oct. 9–11), Society for

the Advancement of Material and Process Engineering, 1984, pp. 136–147.

84A37382

An in situ diffuse reflectance-Fourier transform infrared technique was developed to determine infrared spectra of graphite fiber prepregs as they were being cured. A bismaleimide, an epoxy, and addition polyimide matrix resin prepregs were studied. An experimental polyimide adhesive was also examined. An analysis of the resulting spectra provided basic insights to changes in matrix resin molecular structure which accompanied reactions such as imidization and crosslinking. An endo-exothermal isomerization involving reactive end-caps was confirmed for the addition polyimide prepregs. The results of this study contribute to a fundamental understanding of the processing of composites and adhesives. Such understanding will promote the development of more efficient cure cycles.

395. Young, P. R.; Stein, B. A.; and Chang, A. C.: Resin Characterization in Cured Graphite Fiber Reinforced Composites Using Diffuse Reflectance—FTIR. *Materials and Processes— Continuing Innovations* (Proceedings of the Twenty-Eighth National SAMPE Symposium and Exhibition, Anaheim, Calif., Apr. 12–14), Society for the Advancement of Material and Process Engineering, 1983, pp. 824–837.

84A17162

The feasibility of using diffuse reflectance in combination with Fourier transform infrared spectroscopy to obtain information on cured graphite fiber reinforced polymeric matrix resin composites was investigated. Several graphite/epoxy, polysulfone, and polyimide composites exposed to thermal or radiation environments were examined. An experimental polyimide-sulfone adhesive tape was also studied during processing. In each case, significant changes in resin molecular structure were observed. These changes in molecular structure were correlated with previously observed changes in material properties providing new insights into material behavior.

Material properties

396. *NASA/Aircraft Industry Standard Specification for Graphite Fiber Toughened Thermoset Resin Composite Material.* NASA RP-1142, June 1985.

85N26923#

A standard specification for a selected class of graphite fiber/toughened thermoset resin matrix material was developed through joint NASA/Aircraft Industry effort. This specification was compiled to provide uniform requirements and tests for qualifying prepreg systems and for acceptance of prepreg batches. The specification applies specifically to a class of composite prepreg consisting of unidirectional graphite fibers impregnated with a toughened thermoset resin that produce laminates with service temperatures from −65°F to 200°F when cured at temperatures below or equal to 350°F. The specified prepreg has a fiber areal weight of 145 g/sq m. The specified tests are limited to those required to set minimum standards for the uncured prepreg and cured laminates, and are not intended to provide design allowable properties.

397. Campbell, M. D.; and Burleigh, D. D.: Thermophysical Properties Data on Graphite/Polyimide Composite Materials. *Composites for Extreme Environments*, ASTM, 1982, pp. 54–72.

83A33118

Experimental data for the thermal conductivity, thermal expansion, specific heat, and emittance of laminates of HTS/NR 150B2 and HTS/PMR 15 are presented. Measurements were made over the temperature range 116 K to 588 K.

398. Cushman, J. B.; and McCleskey, S. F.: *Design Allowables Test Program, Celion 3000/ PMR-15 and Celion 6000/PMR-15, Graphite/ Polyimide Composites.* NASA CR-165840, June 1982.

82N30338#

A design allowables test program was conducted on Celion 3000/PMR-15 and Celion 6000/PMR-15 graphite/polyimide composites to establish material performance over a 116 K (−250°F) to 589 K (600°F) temperature range. Effects of aging, thermal cycling, and moisture were also evaluated.

399. Ehret, R. M.; Scanlan, P. R.; and Rosen, C. D.: *Development of Design Allowable Data for Celion 6000/LARC-160, Graphite/Polyimide Composite Laminates.* NASA CR-165985, Nov. 1982.

83N16391#

A design allowables test program was conducted on a Celion 6000/LARC-160 graphite polyimide composite to establish material performance over a 116 K (−250°F) to 589 K (600°F) temperature range. Tension, compression, in-plane shear, and short beam shear properties were determined for uniaxial, quasi-isotropic, and ±45 deg laminates. Effects of thermal aging and moisture saturation on mechanical properties were also evaluated.

400. Ekvall, J. C.; and Griffin, C. F.: Design Allowables for T300/5208 Graphite/Epoxy Composite Materials. *22nd AIAA Structures, Structural Dynamics and Materials Conference* (Atlanta, Ga., Apr. 6–8), *Technical Papers, Part 1*, 1981, pp. 416–422. (Available as AIAA Paper 81-0541.)

81A29421#

Material coupon tests were conducted to formulate design allowables for T300/5208 graphite/epoxy unidirectional tape and bidirectional fabric composites. This paper gives a description of the tests conducted, representative ply level and laminate test data, and results of the statistical analysis. The design allowables for tension and compression strength are given and include the effect of notches, impact damage, temperature extremes, and moisture absorption.

401. Farley, G. L.: *Effects of Static Tensile Load on the Thermal Expansion of Gr/PI Composite Material.* NASA TP-1867, June 1981.

81N24179#

The effect of static tensile load on the thermal expansion of Gr/PI composite material was measured for seven different laminate configurations. A computer program was developed which implements laminate theory in a piecewise linear fashion to predict the coupled nonlinear thermomechanical behavior. Static tensile load significantly affected the thermal expansion characteristics of the laminates tested. This effect is attributed to a fiber instability micromechanical behavior of the constituent materials. Analytical results correlated reasonably well with free thermal expansion tests (no load applied to the specimen). However, correlation was poor for tests with an applied load.

402. Garber, D. P.: *Tensile Stress-Strain Behavior of Graphite/Epoxy Laminates.* NASA CR-3592, Aug. 1982.

82N30568#

The tensile stress-strain behavior of a variety of graphite/epoxy laminates was examined. Longitudinal and transverse specimens from eleven different layups were monotonically loaded in tension to failure. Ultimate strength, ultimate strain, and stress-strain curves were obtained from four replicate tests

90

in each case. Polynomial equations were fitted by the method of least squares to the stress-strain data to determine average curves. Values of Young's modulus and Poisson's ratio, derived from polynomial coefficients, were compared with laminate analysis results.

403. Hanagud, S.; Tayebi, A.; Clinton, R. G.; and Nayak, B. M.: Comparative Evaluation of Woven Graphite-Epoxy Composites. *20th AIAA Structures, Structural Dynamics and Materials Conference* (St. Louis, Mo., Apr. 4–6), *Technical Papers on Dynamics and Loads*, 1979, pp. 275–284. (Available as AIAA Paper 79-0777.)

79A28280#

A comparative evaluation of some of the mechanical properties of woven graphite-epoxy composites is discussed in this report. The types of weaves selected are plain, satin, and tri-directional. The composites made of these fabrics have been compared to composites made from unidirectional tapes under static and fatigue loading.

404. Hashin, Z.; Rosen, B. W.; and Pipes, R. B.: *Nonlinear Effects on Composite Laminate Thermal Expansion.* NASA CR-3088, Feb. 1979.

79N17265#

Analyses of graphite/polyimide laminates have shown that the thermomechanical strains cannot be separated into mechanical strain and free thermal expansion strain. Elastic properties and thermal expansion coefficients of unidirectional graphite/polyimide specimens were measured as a function of temperature to provide inputs for the analysis. The experimental results demonstrated the effects predicted by the analysis, namely, dependence of thermal expansion coefficients on load and anisotropy of thermal expansion under load.

405. Haskins, J. F.; Kerr, J. R.; and Stein, B. A.: Flight Simulation Testing of Advanced Composites for Supersonic Cruise Aircraft Applications. *18th AIAA Structures, Structural Dynamics and Materials Conference and Aircraft Composites: The Emerging Methodology for Structural Assurance* (San Diego, Calif., Mar. 21–25), *Technical Papers, Volume A*, 1977, pp. 236–245. (Available as AIAA Paper 77-401.)

77A25753#

The goal of this program was to characterize advanced composite systems before and after exposures to simulated supersonic cruise environments for real times of up to 50,000 hours. Fatigue data for specimens tested at various temperatures and stress ratios are presented. The changes in properties and the degradation mechanisms during high temperature aging are discussed and illustrated using metallographic and fractographic techniques. Initial results of long term flight simulation tests are presented and compared with wearout model predictions based on accelerated flight simulation exposures.

406. Hyer, M. W.; Herakovich, C. T.; Milkovich, S. M.; and Short, J. S., Jr.: Temperature Dependence of Mechanical and Thermal Expansion Properties of T300/5208 Graphite/Epoxy. *Composites*, vol. 14, July 1983, pp. 276–280. (Presented at the Symposium on Environmental Effects on Fibre-Reinforced Plastics, London, England, July 12–13, 1983.)

83A42811

The 116–394 K temperature range elastic constants, ultimate strengths, and thermal expansion behavior of T300/5208 graphite/epoxy laminates of various layup types have been determined. It is noted that most properties exhibit nearly linear temperature dependencies, although the transverse modulus and axial thermal expansion vary nonlinearly with temperature, and change in value by an order of magnitude over the present temperature range.

407. Hyer, M. W.; and Hagaman, J. A.: The Effects of Thermal Cycling on the Thermal Deformation of Graphite-Polyimide. Paper presented at the Society for Experimental Stress Analysis Spring Meeting, San Francisco, Calif., May 20–25, 1979.

80A10210#

The effects of repetitive thermal cycling on the thermal deformation of graphite-fiber-reinforced polyimide laminates are investigated. Unsymmetric and unidirectional laminate specimens were cycled in an oven between room temperature and 180° or 315°C five times, and temperature and deformation were recorded as a function of time in order to measure thermal deformation.

408. Hyer, M. W.; and Hagaman, J. A.: *Thermal Cycling Graphite-Polyimide.* NASA CR-163294, Apr. 1979.

80N26397#

The effects of repetitive thermal cycling on the temperature-thermal deformation relation of graphite-polyimide were determined. A piecewise linear theory, based on classical lamination theory and using the variation of mechanical and thermal expansion properties with temperature, was compared with the experimental results.

409. Johnson, R. R.; Kural, M. H.; and Mackey, G. B.: *Thermal Expansion Properties of Composite Materials.* NASA CR-165632, July 1981.

81N31576#

Thermal expansion data for several composite materials, including generic epoxy resins, various graphite, boron, and glass fibers, and unidirectional and woven fabric composites in an epoxy matrix, were compiled. A discussion of the design, material, environmental, and fabrication properties affecting thermal expansion behavior is presented. Test methods and their accuracy are discussed. Analytical approaches to predict laminate coefficients of thermal expansion (CTE) based on lamination theory and micromechanics are also included. A discussion is included of methods of tuning a laminate to obtain a near-zero CTE for space applications.

410. Kerr, J. R.; and Haskins, J. F.: *Time-Temperature-Stress Capabilities of Composite Materials for Advanced Supersonic Technology Application, Phase 1.* NASA CR-159267, Apr. 1980.

80N33496#

Implementation of metal and resin matrix composites into supersonic vehicle usage is contingent upon accelerating the demonstration of service capacity and design technology. Because of the added material complexity and lack of extensive service data, laboratory replication of the flight service will provide the most rapid method of documenting the airworthiness of advanced composite systems. A program in progress to determine the time-temperature-stress capabilities of several high temperature composite materials includes thermal aging, environmental aging, fatigue, creep, fracture, and tensile tests as well as real time flight simulation exposure. The program has two parts. The first includes all the material property determinations and aging and simulation exposures up through 10,000 hours. The second continues these tests up to 50,000 cumulative hours. Results are presented of the 10,000 hour phase, which has now been completed.

411. Kerr, J. R.; and Haskins, J. F.: Effects of 50,000 Hours of Thermal Aging on Graphite/Epoxy

and Graphite/Polyimide Composites. *23rd AIAA Structures, Structural Dynamics and Materials Conference* (New Orleans, La., May 10–12), *Collection of Technical Papers, Part 1,* 1982, pp. 101–108. (Available as AIAA Paper 82-0657.)

82A30087#

Thermal effects on tensile strengths of advanced composite systems have been determined for exposure time of 100 to 50,000 hours (5.7 years). Exposures were conducted at both ambient and reduced pressures at two temperatures for each composite. At the completion of the various aging periods, specimens were removed from the specially constructed aging furnaces, visually examined, and tensile tested at elevated temperature. After tensile testing, many of the thermal aging specimens were examined using a scanning electron microscope. Results of these studies are presented, and the changes in properties and the degradation mechanisms during high-temperature aging are discussed and illustrated using metallographic techniques.

412. Kulkarni, S. V.; Rosen, B. W.; and Rice, J. S.: An Investigation of the Compressive Strength of Kevlar 49/Epoxy Composites. *Composites,* vol. 6, Sept. 1975, pp. 217–225.

75A43289

Tests were performed to evaluate the effect of a wide range of variables including matrix properties, interface properties, fiber prestressing, secondary reinforcement, and others on the ultimate compressive strength of Kevlar 49/epoxy composites. In addition, a theoretical study was conducted to determine the influence of fiber anisotropy and lack of perfect bond between fiber and matrix on the shear mode microbuckling.

413. McCleskey, S. F.; Cushman, J. B.; and Skournal, D. E.: High Temperature Composites for Advanced Missile and Space Transportation Systems. *23rd AIAA Structures, Structural Dynamics and Materials Conference* (New Orleans, La., May 10–12), *Collection of Technical Papers, Part 1,* 1982, pp. 212–222. (Available as AIAA Paper 82-0707.)

82A30098#

A study has been conducted to characterize a state-of-the-art graphite/polyimide composite system by determining mechanical and thermophysical properties of selected laminates over a temperature range of −250°F to 600°F. The material studied was Celion 3000/PMR-15. Material property data

obtained from testing included tension, compression and shear strengths, and coefficient of thermal expansion.

414. McMahon, P. E.; and Ying, L.: *Effects of Fiber/Matrix Interactions on the Properties of Graphite/Epoxy Composites.* NASA CR-3607, Sept. 1982.

83N10120#

Efforts were made to design experiments to study the wetting behavior of carbon fibers with various finish variants and their effect on adhesion joint strength. The properties of composites with various fiber finishes were measured and compared to the base-line properties of a control. It was shown that by tailoring the interphase properties, a 30% increase in impact toughness was achieved without loss of mechanical properties at both room and elevated temperatures.

415. Milkovich, S. M.; and Herakovich, C. T.: *Temperature Dependence of Elastic and Strength Properties of T300/5208 Graphite-Epoxy.* NASA CR-173616, Apr. 1984.

84N25767#

Experimental results are presented for the elastic and strength properties of T300/5208 graphite-epoxy at room temperature, 116 K (−250°F), and 394 K (+250°F). Results are presented for unidirectional 0, 90, and 45 degree laminates, and angle-ply laminates.

416. Morris, D. H.; and Simonds, R. A.: The Effect of Extreme Temperatures on the Elastic Properties and Fracture Behavior of Graphite/Polyimide Composites. *Mechanical Behavior of Materials—IV* (Proceedings of the Fourth International Conference, Stockholm, Sweden, Aug. 15–19, 1983), *Volume 1,* Pergamon Press, 1984, pp. 517–523.

84A45362

The influence of elevated and cryogenic temperatures on the elastic moduli and fracture strengths of several C6000/PMR-15 and C6000/NR-15082 laminates was measured. Tests were conducted at −157°C, 24°C, and 316°C (−250°F, 75°F, and 600°F). Both notched and unnotched laminates were tested. The average stress failure criterion was used to predict the fracture strength of quasi-isotropic notched laminates.

417. Reese, C.; and Sorem, J., Jr.: *Statistical Distribution of Mechanical Properties for Three Graphite-Epoxy Material Systems.* NASA CR-165736, July 1981.

81N30187#

This investigation was concerned with tension, compression, and short beam shear coupon testing of large samples from three different material suppliers to determine their statistical strength behavior. Statistical results indicate that a two parameter Weibull distribution model provides better overall characterization of material behavior for the graphite-epoxy systems tested than does the standard normal distribtuion model that is employed for most design work.

418. Rogers, A. K.; Tajima, Y. A.; and Young, R. C.: Material Characterization and Specification Development for 350°F Curing Epoxy-Graphite Materials. *Proceedings of the 27th National SAMPE Symposium and Exhibition* (San Diego, Calif., May 4–6), Society for the Advancement of Material and Process Engineering, 1982, pp. 277–291.

83A20443

New analytical techniques and a processability laminate have been used in the development of a material specification for 350°F curing epoxy-graphite materials. The development of the specification required the evaluation of prepregs which had variations in their processing behavior that were detected only in production. Chemical techniques were used to analyze the prepregs and establish limits on the concentrations of the various chemical constituents. A batch acceptance test laminate was designed to determine suitability of prepregs for production usage. The new specification requires that the processability laminate be ultrasonically and micrographically inspected for voids and porosity.

419. Tompkins, S. S.: *Thermal Expansion of Selected Graphite Reinforced Polyimide-, Epoxy-, and Glass-Matrix Composites.* NASA TM-87572, July 1985.

85N32149#

The thermal expansion of three epoxy-matrix composites, a polyimide-matrix composite, and a borosilicate glass-matrix composite, each reinforced with continuous carbon fibers, has been measured and compared.

420. Tompkins, S. S.; and Williams, S. L.: Effects of Thermal Cycling on Residual Mechanical Properties of C6000/PMR-15 Graphite Polyimide. *23rd AIAA Structures, Structural Dynamics and Materials Conference* (New Orleans, La., May 10–12), *Collection of Technical Papers, Part 1,* 1982, pp. 239–246. (Available as AIAA Paper 82-0710.)

82A30101#

An experimental study to determine the effects of thermal cycling on residual mechanical properties of graphite/polyimide composite has been conducted. Interlaminar shear, flexure, and compression strengths were measured at room temperature and 316°C on unidirectional and quasi-isotropic laminates subjected to 150 and 500 thermal cycles between −156°C and 316°C.

421. Wakelyn, N. T.: On the Structure of Poly(Etheretherketone) (PEEK). *Polymer Communications*, vol. 25, Oct. 1984, pp. 306–308.

85A41318

Annealed, hot pressed, and melt-quenched PEEK specimens have been analyzed by wide-angle X-ray diffraction. The analysis suggests, in part, that structural studies based on the reported similarity between the structure of PEEK and that reported for poly(p-phenylene oxide) are valid and may be extended to a structure more in keeping with the chemical structure of the material.

422. Walrath, D. E.; and Adams, D. F.: *Iosipescu Shear Properties of Graphite Fabric/Epoxy Composite Laminates.* NASA CR-176316, June 1985.

86N12258#

The Iosipescu shear test method was used to measure the in-plane and interlaminar shear properties of four T300 graphite fabric/934 epoxy composite materials. Different weave geometries tested include an Oxford weave, a 5-harness satin weave, an 8-harness satin weave, and a plain weave with auxiliary warp yarns. Both orthogonal and quasi-isotropic layup laminates were tested. In-plane and interlaminar shear properties were obtained for laminates of all four fabric types.

423. Zimmerman, R. S.; and Adams, D. F.: *Mechanical Properties Testing of Candidate Polymer Matrix Materials for Use in High Performance Composites.* NASA CR-177970, Dec. 1985.

86N15349#

The mechanical properties of four candidate neat resin systems for use in graphite/epoxy composites are characterized. This includes tensile and shear stiffnesses and strengths, coefficients of thermal and moisture expansion, and fracture toughness. The neat resins tested are Hexcel HX-1504, Narmco 5245-C, American Cyanamid CYCOM 907, and Union Carbide ERX-4901A (MDA). Results are compared with those obtained for four other epoxy resins tested in a prior program, i.e., Hercules 3502, 2220-1, and 2220-3, and Ciba-Geigy Fibredux 914, as well as with available Hercules 3501-6 data.

424. Zimmerman, R. S.; Adams, D. F.; and Walrath, D. E.: *Investigation of the Relations Between Neat Resin and Advanced Composite Mechanical Properties.* NASA CR-172303, Nov. 1984.

85N13911#

A detailed evaluation of one untoughened epoxy baseline resin and three toughened epoxy resin systems was performed. The Hercules 3502, 2220-1, and 2220-3, and Ciba-Geigy Fibredux 914 resin systems were supplied in the uncured state by NASA-Langley and cast into thin flat specimens and round dogbone specimens. Tensile and torsional shear measurements were performed at three temperatures and two moisture conditions. Coefficients of thermal expansion and moisture expansion were also measured. Extensive scanning electron microscopic examination of fracture surfaces was performed, to permit the correlation of observed failure modes with the environmental conditions under which the various specimens were tested. A micromechanics analysis was used to predict the unidirectional composite response under the various test conditions using the neat resin experimental results as the required input data.

Test methods

425. *Standard Tests for Toughened Resin Composites, Revised Edition.* NASA RP-1092 Rev., July 1983.

83N31730#

Several toughened resin systems are evaluated to achieve commonality for certain kinds of tests used to characterize toughened resin composites. Specifications for five tests were standardized; these test standards are described.

426. Bowles, D. E.; Tenney, D. R.; Post, D.; and Herakovich, C. T.: Moiré Interferometry for Thermal Expansion of Composites. *Experimental Mechanics*, vol. 21, Dec. 1981, pp. 441–447. (Presented at the Society for Experimental Stress Analysis Spring Meeting, Dearborn, Mich., May 31–June 4, 1981.)

82A16995

Moiré interferometry by reflection has been demonstrated using a real reference grating of 1200 lines/mm. The method is shown to be well adapted to thermal environments. Thermal expansion coefficients of graphite-epoxy composites have been measured with high precision over a wide range from nearly zero to 3300 microstrains in the temperature range of 297–422 K. Random errors characterized by one standard deviation can be as small as one microstrain.

427. Bowles, D. E.; Post, D.; Herakovich, C. T.; and Tenney, D. R.: *Thermal Expansion of Composites Using Moiré Interferometry.* NASA TM-80788, Aug. 1980.

80N29429#

An experimental technique for precise measurement of the thermal response of fiber-reinforced composite material uses Moiré interferometry with fringe multiplication which yields a sensitivity of 833 nm (32.8 μin.) per fringe. Results from the technique are compared with those obtained from electrical resistance strain gages, and also those predicted from classical lamination theory.

428. Brooks, E. W., Jr.; Herakovich, C. T.; Post, D.; and Hyer, M. W.: *Advances in Moiré Interferometry for Thermal Response of Composites.* NASA CR-169121, Mar. 1982.

82N27410#

An experimental technique for the precise measurement of the thermal response of both sides of a laminated composite coupon specimen uses Moiré interferometry with fringe multiplication which yields a sensitivity of 833 nm (32.8 μin.) per fringe. The technique is shown to be capable of producing the sensitivity and accuracy necessary to measure a wide range of thermal responses.

429. Bush, H. G.; and Weller, T.: *A Biaxial Method for Inplane Shear Testing.* NASA TM-74070, Apr. 1978.

78N21489#

A biaxial method for performing inplane shear tests of materials using a shear frame is described. Aluminum plate and sandwich specimens were used to characterize the uniformity of shear strain imparted by the biaxial method of loading as opposed to the uniaxial method. The inplane stiffening effect of aluminum honeycomb core was determined. Test results for (±45) graphite-epoxy laminates are presented. Some theoretical considerations of subjecting an anisotropic material to a uniform shear deformation are discussed.

430. Chen, J. S.; and Hunter, A. B.: *Development of Quality Assurance Methods for Epoxy-Graphite Prepreg.* NASA CR-3531, Mar. 1982.

82N22318#

Quality assurance methods for graphite-epoxy prepregs were developed. Liquid chromatography, differential scanning calorimetry, and gel permeation chromatography were investigated. The chromatography and calorimetry techniques were all successfully developed as quality assurance methods for graphite-epoxy prepregs. The liquid chromatography method was the most sensitive to changes in resin formulation.

431. Clark, R. K.; and Lisagor, W. B.: *Compression Testing of Graphite/Epoxy Composite Materials. Test Methods and Design Allowables for Fibrous Composites* (Proceedings of the Symposium, Dearborn, Mich., Oct. 2–3, 1979), ASTM, 1981, pp. 34–53.

81A47804

Three methods for compression testing coupons of filament-reinforced polymer-matrix composite materials are evaluated, to identify the sensitivity of the test techniques to laminate, specimen, and test parameters. A "wedge-grip" compression fixture, a face-supported compression fixture, and an end-loaded coupon fixture are described. Specimens of 12.5-, 25-, and 50-mm widths, 8-, 16-, and 24-ply thicknesses, and of various fiber orientations were used to test the "wedge-grip" compression fixture and the face-supported fixture; the end-loaded coupon fixture was tested using 16-ply specimens. Compressive strain-stress, strength, and modulus data obtained with the fixtures and evaluations showing the effects of all the test parameters are presented. One of the conclusions asserts that the "wedge-grip" compression fixture provides good stress-strain data to failure for unidirectional and quasi-isotropic laminates.

432. Claus, R. O.; Zewekh, P. S.; Turner, T. M.; Wade, J. C.; Rogers, R. T.; and Garg, A. O.: *Optical Fiber Interferometer for the Study of Ultrasonic Waves in Composite Materials.* NASA CR-164916, Oct. 1980.

82N11121#

The possibility of acoustic emission detection in composites using embedded optical fibers as sensing elements was investigated. Optical fiber interferometry, fiber acoustic sensitivity, fiber interferometer calibration, and acoustic emission detection are reported.

433. Czarnek, R.; and Post, D.: Edge Effects in Composites by Moiré Interferometry. *Experimental Techniques*, vol. 7, Jan. 1983, pp. 18–21.

83A28076

The very high sensitivity of Moiré interferometry has permitted the present edge effect experiments to be conducted at a low average stress and strain level, assuring linear and elastic behavior in the composite material samples tested. Sensitivity corresponding to 2450 line/mm Moiré was achieved with a 0.408 micron/fringe. Simultaneous observations of the specimen face and edge displacement fields showed good fringe definition despite the 1-mm thickness of the specimens and the high gradients, and it is noted that the use of a carrier pattern and optical filtering was effective in even these conditions. Edge effects and dramatic displacement gradients were confirmed in angle-ply composite laminates.

434. Daniel, I. M.; Niiro, T.; and Koller, G. M.: *Development of Orthotropic Birefringent Materials for Photoelastic Stress Analysis.* NASA CR-165709, May 1981.

81N26183

Materials were selected and fabrication procedures developed for orthotropic birefringent materials. An epoxy resin (Maraset 658/558 system) was selected as the matrix material. Fibers obtained from style 3733 glass cloth and type 1062 glass roving were used as reinforcement. Unidirectional, angle-ply, and quasi-isotropic laminates of two thicknesses and with embedded flaws were fabricated. The matrix and the unidirectional glass/epoxy material were fully characterized. The density, fiber volume ratio, mechanical, and optical properties were determined.

435. Garcia, R.; McWithey, R. R.; and Weisshaar, T. A.: An Experimental and Analytical Investigation of the Rail Shear-Test Method as Applied to Composite Materials. Paper presented at the Society for Experimental Stress Analysis Spring Meeting, San Francisco, Calif., May 20–25, 1979.

80A10208#

This report presents the results from an experimental and analytical investigation of the stress distributions occurring in a rail shear test. The effects of non-uniform stresses induced by differential thermal expansion, rail flexibility, and specimen aspect ratio on measured shear modulus and ultimate strength of composite laminates are shown. A two-dimensional linearly elastic finite element model was used to analytically determine how various geometric parameters influenced the magnitude and distribution of inplane normal and shear stresses.

436. Hagemaier, D. J.: NDT of DC-10 Graphite-Epoxy Rudder. *Materials Evaluation*, vol. 36, May 1978, pp. 57–61.

78A35445

The results of visual and ultrasonic nondestructive tests (NDT) on a DC-10 rudder constructed of Thornel 300/5208 graphite-epoxy are presented. A viscous couplant was used for both contact pulse-echo ultrasonics and the Fokker bondtester. Attention is given to the stiffness of cohesive bond structures and to possible delamination.

437. Harris, C. E.; and Morris, D. H.: On the Use of the Three Point Bend and Compact Tension Specimens To Measure Fracture Toughness of Composite Laminates. *Mechanical Behavior of Materials—IV* (Proceedings of the Fourth International Conference, Stockholm, Sweden, Aug. 15–19, 1983), *Volume 1,* Pergamon Press, 1984, pp. 549–555.

84A45365

The influence of specimen thickness on the fracture toughness of two laminates and three specimen geometries was investigated. As thickness increased the toughness decreased and approached an asymptotic value that was dependent upon the type of laminate but was practically independent of specimen geometry. Enhanced X-ray photographs and removal of an outside ply revealed that most of the delaminations were surface effects.

438. Hayford, D. T.; and Henneke, E. G., II: A Model for Correlating Damage and Ultrasonic Attenuation in Composites. *Composite Materials: Testing and Design* (Proceedings of the Fifth Conference, New Orleans, La., Mar. 20–22, 1978), ASTM, 1979, pp. 184–200.

80A21139

Ultrasonic attenuation data measured by the pulse-echo technique have been obtained versus load for two graphite-epoxy laminates and are presented. The attenuation-load data display distinct changes at load values associated with the initiation of transverse cracks in the weakest plies and with the knee in the bilinear stress-strain curve.

439. Hayford, D. T.; Henneke, E. G., II; and Stinchcomb, W. W.: The Correlation of Ultrasonic Attenuation and Shear Strength in Graphite-Polyimide Composites. *Journal of Composite Materials,* vol. 11, Oct. 1977, pp. 429–444.

78A17792

The buffer rod technique for measuring attenuation in thin specimens is modified here to apply to specimens having intermediate thicknesses and high attenuation. The described procedure, which requires only one accessible surface of the material, was used to determine the initial attenuation values of ultrasonic waves in short beam shear specimens of graphite-polyimide composite material. It is shown that there is good correlation between the initial attenuation values and the shear strengths of the specimens determined by the standard short beam shear test method.

440. Hayford, D. T.; and Henneke, E. G.: *Theory of Ultrasonic Diffraction by Damage Developed in Thin Laminated Composites.* NASA CR-155161, Oct. 1977.

77N33266#

The apparent attenuation which would result if certain damage states (transverse cracks and delaminations) are introduced into a graphite/epoxy laminate is investigated. Experimental data for two different laminates are presented which show changes in the apparent attenuation of about 1 dB. These changes generally occur at loads which correspond to the range predicted for the formation of the damage. The predicted changes in the attenuation for several simple and common damage states are well within the range of experimental values.

441. Herakovich, C. T.; Post, D.; and Nemeth, M. P.: On the Off-Axis Tension Test for Unidirectional Composites. *Composites Technology Review,* vol. 5, Summer 1983, pp. 61–68.

83A41033

The off-axis tensile test was examined experimentally to obtain actual displacement fields over the surface of graphite/polyimide coupon specimens. The experimental results were compared with approximate analytical solutions. An optical method of high sensitivity Moiré interferometry was used to determine the actual displacements to high precision.

442. Heyman, J. S.; and Cantrell, J. H., Jr.: Application of an Ultrasonic Phase Insensitive Receiver to Material Measurements. Paper presented at the Institute of Electrical and Electronics Engineers Ultrasonics Symposium, Phoenix, Ariz., Oct. 26–28, 1977.

78A18794

The theory of a phase insensitive receiver based on acousto-electric effect is presented along with experimental characteristics of a CdS acousto-electric converter (AEC). Since the AEC is nearly phase insensitive, it is ideal for measurements in inhomogeneous materials and/or materials with irregular flatness and parallelism. Through transmission, ultrasonic C-scan data of phantom flaws demonstrates a significant improvement in flaw characterization with an AEC over that of a conventional transducer. In addition, measurements with conventional transducers in anisotropically stressed metal samples are shown to lead to grossly inaccurate results due to phase sensitivity. Various other measurements are presented with data contrasting conventional transducers with an AEC in specific NDE applications.

443. Heyman, J. S.; and Cantrell, J. H., Jr.: Effects of Material Inhomogeneities on Ultrasonic Measurements—The Problem and a Solution. *Nondestructive Evaluation and Flaw Criticality for Composite Materials* (Proceedings of the Symposium, Philadelphia, Pa., Oct. 10–11, 1978), ASTM, 1979, pp. 45–56.

80A33379

Most ultrasonic measurements of materials involve the generation of an acoustic wave and the propagation of that wave from a transducer through a coupling medium to a specimen under test. After interacting with the specimen, the wave propagates through the coupling medium to a receiving transducer and is converted to an electrical signal. In this paper, the role that the receiving transducer plays in ultrasonic measurements is examined. The phase-sensitive nature of conventional receiving transducers has, for the most part, been neglected in nondestructive evaluations. This is shown to lead to significant data misinterpretation. A new acoustoelectric transducer (AET) has been developed which is phase insensitive. Comparative data obtained with both conventional and AET transducers are presented and discussed. The AET is shown to produce more accurate measurements for the cases investigated.

444. Hurwitz, F. I.; and Behrendt, D. R.: Application of Iosipescu Specimen Geometry to Determination of Shear Strength in Unidirectional Composites. *Proceedings of the 16th National Technical Conference* (Albuquerque, N. Mex., Oct. 9–11), Society for the Advancement of Material and Process Engineering, 1984, pp. 609–620.

85A37402

The Iosipescu (1967) test method, in conjunction with a finite element analysis of anisotropic, unidirectionally reinforced composite specimens, can be used to derive composite shear strength values despite the small size of the samples used. When such a specimen is loaded parallel to the fiber direction, fracture occurs by shear within a narrow region near the center of the specimen; this is consistent with the maximum shear area determined by finite element analysis. The resulting fracture surface, as observed, is characteristic of shear failure, occurring either within the matrix or at the fiber-matrix interface.

445. Hyer, M. W.; and Liu, D.: *Stresses in Pin-Loaded Orthotropic Plates Using Photoelasticity.* NASA CR-172498, Oct. 1984.

85N16204#

The stresses in transparent glass-epoxy plates loaded by a steel pin through a hole were determined by photoelasticity. The stresses around the hole edge, across the net section, along the shear out line, and on the centerline below the hole for quasi-isotropic, unidirectional, and angle ply plates are outlined. Stresses in an isotropic comparison specimen are also presented. Stress concentration factors for several locations around the plates are tabulated. The experimental apparatus and the experimental technique are discussed. The isochromatic and isoclinic fringe patterns for the four plates are shown. A description of the necessary photoelastic theory is appended.

446. Hyer, M. W.; and Liu, D. H.: *An Assessment of the Accuracy of Orthotropic Photoelasticity.* NASA CR-3773, Mar. 1984.

84N19924#

The accuracy of orthotropic photoelasticity was studied. The study consisted of both theoretical and experimental phases. In the theoretical phase a stress-optic law was developed. The stress-optic law included the effects of residual birefringence in the relation between applied stress and the material's optical response. The experimental phase had several portions. First, it was shown that four-point bending tests and the concept of an optical neutral axis could be conveniently used to calibrate the stress-optic behavior of the material. Second, the actual optical response of an orthotropic disk in diametral compression was compared with theoretical predictions. Third, the stresses in the disk were determined from the observed optical response, the stress-optical law, and a finite-difference form of the plane stress equilibrium equations. It was concluded that orthotropic

photoelasticity is not as accurate as isotropic photoelasticity. This is believed to be due to the lack of good fringe resolution and the low sensitivity of most orthotropic photoelastic materials.

447. Hyer, M. W.; Herakovich, C. T.; and Post, D.: Thermal Expansion of Graphite Epoxy. *1982 Advances in Aerospace Structures and Materials* (Proceedings of the Winter Annual Meeting, Phoenix, Ariz., Nov. 14–19), ASME, 1982, pp. 107–113.

83A27443#

Two optical techniques to measure the thermal expansion of fiber-reinforced composites are discussed. These techniques are Moiré interferometry and Fizeau interferometry. The pertinent features of each setup are presented and the preparation of the composite specimens discussed. Thermal expansion measurements in the range 116 to 422 K for 0 deg, 90 deg, and quasi-isotropic graphite-epoxy composite laminates are presented.

448. Johnston, N. H.; O'Brien, T. K.; Morris, D. H.; and Simonds, R. A.: Interlaminar Fracture Toughness of Composites. II—Refinement of the Edge Delamination Test and Application to Thermoplastics. *Materials and Processes—Continuing Innovations* (Proceedings of the Twenty-Eighth National SAMPE Symposium and Exhibition, Anaheim, Calif., Apr. 12–14), Society for the Advancement of Material and Process Engineering, 1983, pp. 502–517.

84A17136

The mixed mode interlaminar fracture toughness is obtained for the two thermoplastic matrices UDEL P1700 polysulfone and ULTEM polyetherimide by means of edge delamination tensile (EDT) tests on unnotched, eleven-ply graphite fiber reinforced composite specimens. A novel method is used to obtain the stiffness parameter employed in the closed form equation, decreasing the number of stiffness measurements required and simplifying the calculations.

449. Kishoni, D.: Application of Digital Pulse Shaping by Least-Squares Method to Ultrasonic Signals in Composites. Paper presented at Review of Progress in Quantitative NDE, Williamsburg, Va., June 23–28, 1985.

86A17828#

For ultrasonic testing in composites, a signal processing tool is identified that can significantly enhance the sharpness of ultrasonic waveforms and provide clearer pictures of the nature of the material

flaw. The technique artificially improves the resolution of the system to discrete events by pulse shaping the measured waveform based on the signal from a reference. The optimum pulse shape operator is determined from a least-squares method in the z-domain.

450. Murri, G. B.; and O'Brien, T. K.: *Interlaminar G(IIc) Evaluation of Toughened-Resin Matrix Composites Using the End-Notched Flexure Test. 26th AIAA Structures, Structural Dynamics and Materials Conference* (Orlando, Fla., Apr. 15–17), *Technical Papers, Part 1*, 1985, pp. 197–202. (Available as AIAA Paper 85-0647.)

85A30247#

The usefulness of the end-notched flexure (ENF) test for measuring G(IIc) for toughened composites was evaluated. Calculations were made using the measured flexural compliance from the ENF test and the measured axial modulus. Tests were conducted on three different composite materials, using laminates with two different span lengths and insert sizes. The error associated with neglecting nonlinear terms in the displacement-curvature relationship was analyzed to demonstrate the influence of specimen geometry on G(IIc) measurement. It is concluded that G(IIc) measurement may vary with the method of calculation, span length, or insert thickness, but these effects are minor compared to the large differences in measurements obtained for different materials. Hence, the ENF test is a valid one for screening materials for improvements in interlaminar shear fracture toughness.

451. Nemeth, M. P.; Herakovich, C. T.; and Post, D.: *On the Off-Axis Tensile Test of Unidirectional Composites.* NASA CR-168853, Apr. 1982.

82N23553#

The off-axis tensile test was examined experimentally to obtain actual displacement fields over the surface of graphite/polyimide coupon specimens. The experimental results were compared with approximate analytical solutions and generated finite element results. An optical method of high sensitivity Moiré interferometry was used to determine the actual displacements to high precision. The approximate analytical solution and the finite element results compare very favorably with the measured centerline displacements in the test section, and the finite element displacement fields provide excellent agreement with the Moiré displacements throughout the specimen.

452. O'Brien, T. K.; Johnston, N. J.; Raju, I. S.; Morris, D. H.; and Simonds, R. A.: *Comparisons of Various Configurations of the Edge Delamination Test for Interlaminar Fracture Toughness.* NASA TM-86433, July 1985.

85N32151#

Various configurations of edge delamination tension test specimens, of both brittle (T300/5208) and toughened-matrix (T300/BP907) graphite reinforced composite laminates, were manufactured and tested.

453. O'Brien, T. K.; Johnston, N. J.; Morris, D. H.; and Simonds, R. A.: *A Simple Test for the Interlaminar Fracture Toughness of Composites. SAMPE Journal*, vol. 18, July–Aug. 1982, pp. 8–15.

82A39473

A simple test method for the measurement of delamination resistance is assessed through its application to 11-ply graphite-epoxy laminates. Comparison of these results with those of width-tapered double cantilever beam (WTDCB) test data shows a similar strain energy release rate, establishing the accuracy of the novel method. Both tests are found to be needed, however, for the quantitative characterization of interlaminar fracture toughness.

454. Phelps, M. L.: *In-Service Inspection Methods for Graphite-Epoxy Structures on Commercial Transport Aircraft.* NASA CR-165746, Nov. 1981.

82N12142#

In-service inspection methods for graphite-epoxy composite structures on commercial transport aircraft are determined. Graphite/epoxy structures, service incurred defects, current inspection practices and concerns of the airline and manufacturers, and other related information were determined by survey. Based on this information, applicable nondestructive inspection methods are evaluated and inspection techniques determined.

455. Raju, B. B.; Camarda, C. J.; and Cooper, P. A.: *Elevated-Temperature Application of the IITRI Compression Test Fixture for Graphite/Polyimide Filamentary Composites.* NASA TP-1496, Sept. 1979.

79N32277#

Seventy-nine graphite/polyimide compression specimens were tested to investigate experimentally the IITRI test method for determining compressive properties of composite materials at room and elevated temperatures (589 K (600°F)). Minor modifications were made to the standard IITRI fixture and

a high degree of precision was maintained in specimen fabrication and load alignment.

456. Reifsnieder, K. L.: Automated Time-Resolved Data Acquisition During Static and Dynamic Loading of Composite Materials. *VDI-Berichte*, no. 313, 1978, pp. 699–703. (Presented at 6th Verein Deutscher Ingenieure, Internationale Konferenz ueber experimentelle Spannungsanalyse, Munich, West Germany, Sept. 18–22, 1978.)

79A36028

This paper reports the development of two new nondestructive techniques for precise determination of the nature and extent of complex damage and the consequent redistribution of stress in composite materials during quasi-static and cyclic or dynamic loading. The two methods incorporate video-thermography and four-component tensor elastic modulus determination. The experimental and analytical aspects of these two methods are discussed and data are presented.

457. Reiss, R.; Clark, R. K.; and Yao, T.-M.: Effect of Load Introduction in Compression Testing of Composite Laminates. *Compression Testing of Homogeneous Materials and Composites* (Proceedings of the Symposium, Williamsburg, Va., Mar. 10–11, 1982), ASTM, 1983, pp. 200–220.

84A29897

Compression testing of composite materials is affected by the manner in which the compressive load is introduced. Two such effects are studied in this paper: (a) the constrained edge effect, in which transverse expansion of the edges is prevented while the axial load is introduced, and (b) nonuniform gripping, as manifested by inplane bending of the test specimen. The principle of minimum complementary energy is used to develop an analytical model that quantifies these two effects upon the measured elastic properties of laminated composites. Numerical results are presented for selected high-strength graphite/epoxy composites.

458. Russell, S. S.; and Henneke, E. G., II: *Signature Analysis of Acoustic Emission From Graphite/Epoxy Composites*. NASA CR-153414, Sept. 1977.

77N30179#

Acoustic emissions were monitored for crack extension across and parallel to the fibers in a single ply and multiply laminates of graphite/epoxy composites. Spectrum analysis was performed on the transient signal to ascertain if the fracture mode can be

characterized by a particular spectral pattern. The specimens were loaded to failure in a tensile machine. Visual observations were made via either an optical microscope or a television camera. The results indicate that several types of characteristics in the time and frequency domain correspond to different types of failure.

459. Short, J. S.; Hyer, M. W.; Bowles, D. E.; and Tompkins, S. S.: *Development of a Priest Interferometer for Measurement of the Thermal Expansion of a Graphite/Epoxy in the Temperature Range 116–366 K*. NASA TM-85176, Sept. 1982.

83N22578#

The thermal expansion behavior of graphite/epoxy laminates between 116 and 366 K was investigated using the Priest interferometer concept. The design, construction, and use of the interferometer, along with the experimental results generated, are described. The experimental program consisted of 25 tests on 25.4 mm and 6.35 mm wide, 8 ply quasi-isotropic T300-5208 graphite/epoxy specimens and 3 tests on a 25.4 mm wide unidirectional specimen. Experimental results are presented for all tests along with a discussion of the interferometer's limitations and some possible improvements in its design.

460. Shuart, M. J.; and Williams, J. G.: Investigating Compression Failure Mechanisms in Composite Laminates With a Transient Fiberglass-Epoxy Birefringent Material. *Experimental Techniques*, vol. 8, Nov. 1984, pp. 24, 25.

85A13932

An experimental study is reported in which a nondestructive technique involving the use of a transparent fiberglass-epoxy composite birefringent material has been used to investigate compression failure mechanisms in graphite-epoxy laminates. It is shown that the birefringency and transparency of the fiberglass-epoxy material permit regions of high stress to be located and the mechanisms of local failure propagation to be identified within the laminate. The material may also be useful for studying stress fields and for identifying failure initiation and propagation mechanisms in a wide variety of composite-structure problems.

461. Shuart, M. J.: An Evaluation of the Sandwich Beam Compression Test Method for Composites. *Test Methods and Design Allowables for Fibrous Composites* (Proceedings of the Symposium, Dearborn, Mich., Oct. 2–3, 1979), ASTM, 1981, pp. 152–165.

81A47809

The sandwich beam in a four-point bending compressive test method for advanced composites is evaluated. Young's modulus and Poisson's ratio were obtained for graphite/polyimide beam specimens tested at 117 K, room temperature, and 589 K. Tensile elastic properties obtained from the specimens were assumed to be equal to the compressive elastic properties and were used in the analysis. Strain gages were used to record strain data. A three-dimensional finite-element model was used to examine the effects of the honeycomb core on measured composite mechanical properties. Results of the analysis led to the following conclusions: (1) a near uniaxial compressive stress state existed in the top cover and essentially all the compressive load was carried by the top cover; (2) laminate orientation, test temperature, and type of honeycomb core material were shown to affect the type of beam failure; and (3) the test method can be used to obtain compressive elastic constants over the temperature range 117 to 589 K.

462. Shuart, M. J.; and Williams, J. G.: *Investigating Compression Failure Mechanisms in Composite Laminates With a Transparent Fiberglass-Epoxy Birefringent Material.* NASA TM-86306, Aug. 1984.

85N11137#

The response and failure of a ±45s class laminate were studied by transparent fiberglass epoxy composite birefringent material. The birefringency property allows the laminate stress distribution to be observed during the test and also after the test if permanent residual stresses occur. The location of initial laminate failure and of the subsequent failure propagation is observed through its transparency characteristics. Experimental results are presented.

463. Singh, J. J.; Holt, W. R.; and Mock, W., Jr.: *Moisture Determination in Composite Materials Using Positron Lifetime Techniques.* NASA TP-1681, July 1980.

80N27428#

A technique was developed which has the potential of providing information on the moisture content as well as its depth in the specimen. This technique was based on the dependence of positron lifetime on the moisture content of the composite specimen. The positron lifetime technique of moisture determination and the results of the initial studies are described.

464. Stiffler, R.; Henneke, E. G., II; and Herakovich, C. T.: Detection of Acoustic Emission From Composite Laminates Using PVF2 Transducers. *Proceedings of the 1st International Symposium on Acoustic Emission From Reinforced Plastics* (San Francisco, Calif., July 19–21), Society of the Plastics Industry, Inc., 1983.

85A29940#

Polyvinylidene fluoride (PVF2), a semicrystalline polymer exhibiting piezoelectricity, is presently used as a sensing transducer in acoustic emission (AE) monitoring of several different composite laminate materials in order to obtain both quasi-static and fatigue loading results. AE signals obtained from PVF2 transducers are compared with those obtained by standard AE sensors. It is noted that PVF2 transducers may, through the application of spectral signal analysis, be able to distinguish between two distinct failure modes which have been observed in two composite laminates of the same material, but employing different lamina stacking sequences.

465. Stiffler, R.: Application of a Peak-Valley Detector for Secant Stiffness Measurements. *Experimental Techniques*, vol. 8, Jan. 1984, pp. 17–20.

84A20374

A technique for monitoring the change in secant stiffness of composite specimens during cyclic-fatigue loading computes the secant modulus every five seconds. A resettable peak-valley detector, which captures and holds the maximum and minimum values of an input analog signal, is used in conjunction with a multichannel data acquisition unit. This detector is also used in conjunction with an analog switch to halt the cycling of the specimen after a certain degradation of stiffness develops for the application of NDE techniques. Results for a T300/5208 graphite-epoxy specimen are presented in the form of a stiffness versus number of cycles curve.

466. Tompkins, S. S.; Bowles, D. E.; and Kennedy, W. R.: A Laser Interferometer Dilatometer for Thermal Expansion Measurements of Composites. *Proceedings of the 5th International Congress on Experimental Mechanics* (Montreal, Canada, June 10–15), Society for Experimental Stress Analysis, 1984, pp. 367–376.

86A13325#

A high precision Fizeau type, laser interferometer dilatometer system has been developed for low-expansion composite materials. The strain resolution is about 1 microstrain. The system is automated to operate over a large temperature range and record data during the test in real time. A technique has been developed to reduce the fringe data in real time to length changes. The dilatometer system is

described and thermal expansion measurements for several fiber-reinforced and particle filled composites are presented.

467. Walrath, D. E.; and Adams, D. F.: *Verification and Application of the Iosipescu Shear Test Method.* NASA CR-174346, June 1984.

85N18376#

Finite element models were used to study the effects of notch angle variations on the stress state within an Iosipescu shear test specimen. These analytical results were also studied to determine the feasibility of using strain gage rosettes and a modified extensometer to measure shear strains in the test specimen. Analytical results indicate that notch angle variations produced only small differences in simulated shear properties. Both strain gage rosettes and the modified extensometer were shown to be feasible shear strain transducers for the test method.

Joints and cutouts

468. Adkins, D. W.; and Pipes, R. B.: End Effects in Scarf Joints. *Composites Science and Technology,* vol. 22, no. 3, 1985, pp. 209–221.

85A28147

Scarf joints with small scarf angles are especially sensitive to stiffness mismatch between adherends and to adherend tip bluntness. Pre-assembly breakage of an adherend tip where it is only a few microns thick can cause significant reduction in joint strength. Mathematically, the reason for such sensitivity is that the solutions to the governing differential equation develop boundary layer character when the scarf angles are small. The boundary layers are regions with large adhesive stresses. Experimental strength data for laminated composite adherends agree with the results of this analysis.

469. Barclay, D. L.: *Design, Fabrication and Test of Graphite/Polyimide Composite Joints and Attachments for Advanced Aerospace Vehicles.* NASA CR-159113, Dec. 1980.

81N25141#

Results of an experimental program to develop several types of graphite/polyimide (GR/PI) bonded and bolted joints for lightly loaded flight components for advanced space transportation systems and high speed aircraft are presented.

470. Crews, J. H., Jr.; Hong, C. S.; and Raju, I. S.: *Stress-Concentration Factors for Finite Orthotropic Laminates With a Pin-Loaded Hole.* NASA TP-1862, May 1981.

81N24178#

Stresses were calculated for finite size orthotropic laminates loaded by a frictionless steel pin in a circular hole of the same diameter. The calculations were based on finite element analyses for six laminates. Stress concentration factors, based on nominal bearing stress, were determined for wide ranges of the ratios of width to diameter w/d and edge distance to diameter e/d.

471. Crews, J. H., Jr.; and Naik, R. A.: Stress Analysis Method for a Clearance-Fit Bolt Under Bearing Loads. *26th AIAA Structures, Structural Dynamics and Materials Conference* (Orlando, Fla., Apr. 15–17), *Technical Papers, Part 1,* 1985, pp. 522–527. (Available as AIAA Paper 85-0746.)

85A30286#

In the present simple analysis method for bolt hole clearance stresses in mechanically fastened joints, an inverse formulation is used with a finite element analysis. Conditions along the bolt hole contact arc are specified by displacement constraint equations. The method was applied to a single fastener clearance-fit joint with typical clearance values, for the cases of a rigid, frictionless bolt and a quasi-isotropic graphite/epoxy laminate. Results indicate that the contact arc, as well as the peak stresses around the hole (and their locations), was strongly influenced by the clearance. After a minor initial nonlinearity, peak stresses varied linearly with applied load. Clearance levels were found to have only a minor influence on overall joint stiffness.

472. Crews, J. H., Jr.: A Survey of Strength Analysis Methods for Laminates With Holes. *Journal of Aeronautical Society of India,* vol. 36, Nov. 1984, pp. 287–303.

85A50067#

The methods for predicting the strength of composite laminates with fastener holes are reviewed for the cases of unloaded as well as loaded holes. Three categories of methods are discussed: (1) the hole-boundary stress (HBS) methods which compare the peak stresses and the laminate strength; (2) the linear elastic fracture methods (LEFM) which assume cracks emanating from the hole and predict failure when the stress intensity factor for a crack equals the laminate fracture toughness; and (3) two different two-parameter (TP) methods: the average-stress

(ASTP) method, which predicts failure when the average stress over a distance from the hole equals the laminate strength; and the point-stress (PSTP) Whitney-Nuismer (1974) approach, which predicts failure when the stress near the hole equals the material strength. The PSTP is relatively accurate and is by far the most widely used of all the prediction methods. However, for large ranges of variables and cases, the current strength predictions may not always be accurate.

473. Crews, J. H., Jr.; and Naik, R. V. A.: *Failure Analysis of a Graphite/Epoxy Laminate Subjected to Bolt Bearing Loads.* NASA TM-86297, Aug. 1984.

84N32436#

Quasi-isotropic graphite/epoxy laminates (T300/5208) were tested under bolt bearing loads to study failure modes, strengths, and failure energy. Specimens had a range of configurations to produce failures by the three nominal failure modes: tension, shearout, and bearing. Radiographs were made after damage onset and after ultimate load to examine the failure modes. Also, the laminate stresses near the bolt hole were calculated for each test specimen configuration and then used with a failure criterion to analyze the test data.

474. Cushman, J. B.; McCleskey, S. F.; and Ward, S. H.: *Test and Analysis of Celion 3000/PMR-15, Graphite/Polyimide Bonded Composite Joints.* NASA CR-3602, Jan. 1983.

83N16787#

Standard single lap, double lap, and symmetric step lap bonded joints of Celion 3000/PMR-15 graphite/polyimide composite were evaluated. Composite-to-composite and composite-to-titanium joints were tested at 116 K (−250°F), 294 K (70°F), and 561 K (550°F). Joint parameters evaluated were lap length, adherend thickness, adherend axial stiffness, lamina stacking sequence, and adherend tapering. Tests of advanced joint concepts were also conducted to establish the change in performance of preformed adherends, scalloped adherends, and hybrid systems. Special tests were conducted to establish material properties of the high temperature adhesive, designated A7F, used for bonding.

475. Cushman, J. B.; McCleskey, S. F.; and Ward, S. H.: *Design, Fabrication and Test of Graphite/Polyimide Composite Joints and Attachments.* NASA CR-3601, Jan. 1983.

83N16786#

The design, analysis, and testing performed to develop four types of graphite/polyimide (Gr/PI) bonded and bolted composite joints for lightly loaded control surfaces on advanced space transportation systems that operate at temperatures up to 561 K (550°F) are summarized.

476. Deiasi, R. J.; and Schulte, R. L.: *Experimental Determination of the Effects of Moisture on Composite-to-Composite Adhesive Joints.* NASA CR-3387, Feb. 1981.

81N17176#

The primary mode of moisture ingress into bonded composite joints is determined using a nuclear probe for deuterium (NPD) to measure the localized D20 content along the length of the adhesive (FM-300 and EA-9601) and through the thickness of bonded composite specimens. Calculated diffusivities and NPD measured equilibrium moisture contents are used to predict the moisture profiles along the length of the adhesives as a function of exposure time, temperature, and relative humidity. These results are compared with the observed moisture profiles to evaluate the extent of enhanced edge diffusion.

477. Duke, J. C., Jr.; Post, D.; Czarnek, R.; and Asundi, A.: *Measurements of Displacement Around Holes in Composite Plates Subjected to Quasi-Static Compression.* NASA CR-169084, May 1982.

82N26712#

Attempts to develop a whole-field, high sensitivity optical technique for measurement of load induced changes of thickness of composite plates are described. Graphite-epoxy plates of quasi-isotropic layup were used as test specimens. Changes of thickness of three plates, each with a central hole of different size, were measured as a function of applied compressive loads.

478. Hart-Smith, L. J.: *Further Developments in the Design and Analysis of Adhesive-Bonded Structural Joints. Joining of Composite Materials* (Proceedings of the Symposium, Minneapolis, Minn., Apr. 16, 1980), ASTM, 1981, pp. 3–31.

82A20978

Previously developed computer programs for analyzing stepped-lap adhesive-bonded joints and doublers have been modified to permit the inclusion of variable adhesives as well as nonuniform adherends. The new version of the analysis program is described with reference to examples illustrating the effects of load redistribution around flaws and porosity.

103

479. Hart-Smith, L. J.: *Bolted Joints in Graphite-Epoxy Composites.* NASA CR-144899, June 1976.

77N15438#

All-graphite-epoxy laminates and hybrid graphite-glass/epoxy laminates were tested. The tests encompassed a range of geometries for each laminate pattern to cover the three basic failure modes—net section tension failure through the bolt hole, bearing, and shearout. Static tensile and compressive loads were applied. The analysis methods developed here for single bolt joints are shown to be capable of predicting the behavior of multi-row joints.

480. Hong, C. S.; and Crews, J. H., Jr.: *Stress-Concentration Factors for Finite Orthotropic Laminates With a Circular Hole and Uniaxial Loading.* NASA TP-1469, May 1979.

79N23140#

Stresses were calculated for finite-width orthotropic laminates with a circular hole and remote uniaxial loading using a two-dimensional finite element analysis with both uniform stress and uniform displacement boundary conditions. Five different laminates were analyzed. Computed results are presented for selected combinations of hole diameter/sheet-width ratio d/w and length-to-width ratio l/w.

481. Hyer, M. W.; and Chastain, P. A.: *Effects of Load Proportioning on the Capacity of Multiple-Hole Composite Joints.* NASA CR-178019, Nov. 1985.

86N15687#

This study addresses the issue of adjusting the proportion of load transmitted by each hole in a multiple-hole joint so that the joint capacity is a maximum. Specifically two-hole-in-series joints are examined. The results indicate that when each hole reacts 50% of the total load, the joint capacity is not a maximum. One hole generally is understressed at joint failure. The algorithm developed to determine the load proportion at each hole which results in maximum capacity is discussed. The algorithm includes two-dimensional finite-element stress analysis and failure criteria. The algorithm is used to study the effects of joint width, hole spacing, and hole to joint-end distance on load proportioning and capacity.

482. Hyer, M. W.; Klang, E. C.; and Cooper, D. E.: *The Effects of Pin Elasticity, Clearance, and Friction on the Stresses in a Pin-Loaded Orthotropic Plate.* NASA CR-175816, Mar. 1985.

85N26927#

The effects of pin elasticity, clearance, and friction on the stresses in a pin-loaded orthotropic plate are studied. The effects are studied by posing the problem as a planar contact elasticity problem, the pin and the plate being two elastic bodies which interact through contact. Coulomb friction is assumed, the pin loads the plate in one of its principal material directions, and the plate is infinite in extent. A collocation scheme and interaction, in conjunction with a complex variable series solution, are used to obtain numerical results.

483. Hyer, M. W.; and Lightfoot, M. C.: Ultimate Strength of High-Load-Capacity Composite Bolted Joints. *Composite Materials: Testing and Design* (Proceedings of the Fifth Conference, New Orleans, La., Mar. 20–22, 1978), ASTM, 1979, pp. 118–136.

80A21135

Presented are the results of a series of tests initiated to obtain baseline data on the load-carrying capacity of bolted joints designed to carry large loads. A total of 100 tests were conducted on three different specimen configurations. The specimens were fabricated from a T300/5208 fiber/resin system in a quasi-isotropic layup. The results presented indicate that for a given ratio of specimen width to hole diameter, the specimens with the smaller holes sustained a higher net-section tensile stress before failure. In addition, for a given ratio of specimen width to hole diameter, the thinner specimens withstood a higher net-section stress. No attempt has been made to correlate the results with theoretical predictions.

484. Hyer, M. W.; and Liu, D. H.: *Use of Two-Dimensional Transmission Photoelastic Models To Study Stresses in Double-Lap Bolted Joints.* NASA CR-165812, Nov. 1981.

82N14533#

The stress distribution in two hole connectors in a double lap joint configuration was studied. The following steps are described: (1) fabrication of photoelastic models of double lap-double hole joints designed to determine the stresses in the inner lap; (2) assessment of the effects of joint geometry on the stresses in the inner lap; and (3) quantification of difference in the stresses near the two holes.

485. Hyer, M. W.; and Liu, D.: Stresses in a Quasi-Isotropic Pin-Loaded Connector Using Photoelasticity. *Experimental Mechanics*, vol. 24, Mar. 1984, pp. 48–53.

84A27430

Birefringent glass-epoxy and a numerical stress separation scheme are used to compute the stresses in the vicinity of a pin-loaded hole. The radial and circumferential stresses at the hole edge and the net section and shear-out stresses are computed. The numerical and experimental results are compared with the computed stresses. The fixture used to load the connector is discussed and typical isochromatic and isoclinic fringe patterns are presented. The stress-separation scheme is briefly discussed.

486. Hyer, M. W.; Perry, J. C.; and Lightfoot, M. C.: Load Transfer in Composite Bolted Joints. *21st AIAA Structures, Structural Dynamics and Materials Conference* (Seattle, Wash., May 12–14), *Technical Papers, Part 2*, 1980, pp. 699–705. (Available as AIAA Paper 80-0779.)

80A35069

The study deals with composite bolted joints, specifically those required to transmit primary loads. Consideration is given to the ultimate load capacity of quasi-isotropic bolted joint specimens as a function of the width of the joint, the diameter of the bolt, the joint thickness, and the number of bolts. Emphasis is placed on the effect of adding a second bolt, in tandem, on the load capacity of the joint.

487. Hyer, M. W.; Lightfoot, M. C.; and Perry, J. C.: *Strain Measurements in Composite Bolted-Joint Specimens.* NASA CR-163296, May 1979.

80N26698#

Strain data from a series of bolted joint tests are presented. Double lap-double hole, double lap-single hole, and open hole tensile specimens were tested and the strain gage locations, load strain responses, and load axial displacement responses are presented.

488. Johnson, W. S.; and Mall, S.: A Fracture Mechanics Approach for Designing Adhesively Bonded Joints. *Delamination and Debonding of Materials*, ASTM, 1985, pp. 189–199.

86A20634

An analytical and experimental investigation was undertaken to determine if the adhesive debond initiation stress could be predicted for arbitrary joint geometries. The analysis was based upon a threshold total strain-energy-release rate concept. Two bonded systems were tested: T300/5208 graphite/epoxy adherends bonded with either EC-3445 or FM-300 adhesive. The approach described herein predicts the maximum stress at which an adhesive joint can be cycled yet not debond.

489. Johnson, R. W.; and McCarty, J. E.: *Design and Fabrication of Graphite-Epoxy Bolted Wing Skin Splice Specimens.* NASA CR-145216, May 1977.

77N30180#

Graphite-epoxy bolted joint specimens were designed and fabricated. These specimens were to be representative of a side-of-body wing skin splice with a 20-year life expectancy in a commercial transport environment. Preliminary tests were performed to determine design values of bearing and net tension stresses. Based upon the information developed, a three-fastener-wide representative wing skin splice was designed for a load of 2627 KN/m (15,000 lbf/in.). One joint specimen was fabricated and tested at NASA. The wing skin splice failed at 106 percent of design ultimate load. This joint design achieved all static load objectives. Fabrication of six specimens, together with their loading fixtures, was completed, and the specimens were delivered to NASA LaRC.

490. Johnson, W. S.; and Mall, S.: *Influence of Interface Ply Orientation on Fatigue Damage of Adhesively Bonded Composite Joints.* NASA TM-86443, June 1985.

85N30365#

An experimental study of cracked-lap-shear specimens was conducted to determine the influence of adherend stacking sequence on debond initiation and damage growth in a composite-to-composite bonded joint. Specimens consisted of quasi-isotropic graphite/epoxy adherends bonded together with either FM-300 or EC 3445 adhesives. The stacking sequence of the adherends was varied such that 0 deg, 45 deg, or 90 deg plies were present at the adherend-adhesive interfaces. Fatigue damage initiated in the adhesive layer in those specimens with 0 deg and 45 deg interface plies. Damage initiated in the form of ply cracking in the strap adherend for the specimens with 90 deg interface plies. The fatigue-damage growth was in the form of delamination within the composite adherends for specimens with the 90 deg and 45 deg plies next to the adhesive, while debonding in the adhesive resulted for the specimens with 0 deg plies next to the adhesive.

491. Klang, E. C.; and Hyer, M. W.: *The Stress Distribution in Pin-Loaded Orthotropic Plates.* NASA CR-176146, June 1985.

85N33547#

The performance of mechanically fastened composite joints was studied. Specifically, a single-bolt connector was modeled as a pin-loaded, infinite plate. The model that was developed used two dimensional, complex variable, elasticity techniques combined with a boundary collocation procedure to produce solutions for the problem. Through iteration, the boundary conditions were satisfied and the stresses in the plate were calculated. Several graphite-epoxy laminates were studied. In addition, parameters such as the pin modulus, coefficient of friction, and pin-plate clearance were varied. Conclusions drawn from this study indicate (1) the material properties (i.e., laminate configuration) of the plate alter the stress state and, for highly orthotropic materials, the contact stress deviates greatly from the cosinusoidal distribution often assumed; (2) friction plays a major role in the distribution of stresses in the plate; (3) reversing the load direction also greatly affects the stress distribution in the plate; (4) clearance (or interference) fits change the contact angle and thus the location of the peak hoop stress; and (5) a rigid pin appears to be a good assumption for typical material systems.

492. Klang, E. C.; and Hyer, M. W.: Damage Initiation at Curved Free Edges—Application to Uniaxially Loaded Plates Containing Holes and Notches. *Recent Advances in Composites in the United States and Japan* (Proceedings of the Symposium, Hampton, Va., June 6–8, 1983), ASTM, 1985, pp. 62–90.

85A46530

The concept which inputs initial free edge interlaminar damage to the differences in material properties among adjacent laminae is presently extended to the case of curved free edges through interpretation in terms of differences in the shear and elongation strains among adjacent laminae. Attention is given to both material property differences and laminate strain spatial distributions at the hole edge. Since a large strain level and a small difference in material properties are found to be as damaging as a small strain and large difference, interlaminar damage will probably begin at a circumferential location other than the net section. These findings are further explored in view of a second extension, using the edges replication technique to record damage at a curved free edge.

493. Knauss, J. F.: *Fatigue Life Prediction of Bonded Primary Joints.* NASA CR-159049, Sept. 1979.

79N31614#

The specific objectives were (1) to ascertain the feasibility of predicting fatigue failure of an adhesive in a primary bonded composite structure by incorporating linear elastic crack growth behavior; and (2) to ascertain if acoustic emission and/or compliance measurement techniques can be used to detect flaws. The validation of a proposed fatigue life prediction methodology was sought through the use of aluminum butt and scarf joint and graphite/epoxy butt joint specimens in a constant amplitude fatigue environment.

494. Mall, S.: *Fatigue Behavior of Adhesively Bonded Joints.* NASA CR-174458, Aug. 1983.

83N36509#

The fatigue damage mechanism of composite-to-composite adhesively bonded joints was characterized. The mechanics of the possible modes of fatigue damage propagation in these joints when subjected to constant amplitude cyclic mechanical loading were investigated.

495. Mall, S.; Johnson, W. S.; and Everett, R. A., Jr.: Cyclic Debonding of Adhesively Bonded Composites. *Adhesive Joints: Formation, Characteristics, and Testing,* Plenum Press, 1984, pp. 639–658.

85A27929

To analyze the fatigue behavior of a simple composite-to-composite bonded joint, a combined experimental and analytical study of the cracked-lap-shear specimen subjected to constant-amplitude cyclic loading was undertaken. Two bonded systems were studied: T300/5208 graphite/epoxy adherends bonded with the adhesives EC 3445 and FM-300. For each bonded system, two specimen geometries were tested: (1) a strap adherend of 16 plies bonded to a lap adherend of 8 plies, and (2) a strap adherend of 8 plies bonded to a lap adherend of 16 plies.

496. Mall, S.; Rezaizadeh, M. A.; and Ramamurthy, G.: *Interaction of Mixed Mode Loading on Cyclic Debonding in Adhesively Bonded Composite Joints.* NASA CR-177990, Oct. 1985.

86N11298#

A combined experimental and analytical investigation of an adhesively bonded composite joint was conducted to characterize the fracture mode dependence of cyclic debonding. The system studied consisted of graphite/epoxy adherends bonded with EC 3445 adhesive. Several types of specimens were tested to provide the cyclic debond growth rate measurements under various load conditions: mode 1,

mixed modes 1 to 2, and mostly mode 2. This study shows that the total strain-energy-release rate is the governing factor for cyclic debonding.

497. Pengra, J. J.; and Wood, R. E.: The Influence of Hole Quality on Graphite Epoxy Composite Laminates. *21st AIAA Structures, Structural Dynamics and Materials Conference* (Seattle, Wash., May 12–14), *Technical Papers, Part 2*, 1980, pp. 687–693. (Available as AIAA Paper 80-0777.)

80A35067#

The paper studies the influence of hole quality on the structural behavior of composite materials. Static and fatigue pin bearing and compression test results are presented and discussed. These results show that a hole chipout defect reduces static and cyclic endurance characteristics.

498. Perry, J. C.; and Hyer, M. W.: *Investigations Into the Mechanical Behavior of Composite Bolted Joints*. NASA CR-163295, June 1979.

80N26664#

Testing procedures and data reduction and interpretation techniques were established for a program to study the mechanical behavior of bolted joints at room temperature, $-157°C$ ($-250°F$), and $315°C$ ($600°F$). The load transfer characteristics, from one bolt to another, in double-bolt joints were investigated by examining data generated in previous investigations. From the results, it appears the increase in load-carrying capacity by adding a second bolt in tandem can be predicted.

499. Reed, D. L.; and Eisenmann, J. R.: Reliability Aspects of a Composite Bolted Scarf Joint. *Composite Reliability* (Proceedings of the Symposium, Las Vegas, Nev., Apr. 15–16, 1974), ASTM, 1975, pp. 105–118.

76A16557

The design, fabrication, static test, and fatigue test of both tension and compression graphite-epoxy candidates for a wing splice representative of a next-generation transport aircraft were the objectives of the reported research program. A single-scarf bolted joint was selected as the design concept. Test specimens were designed and fabricated to represent an upper-surface and a lower-surface panel containing the splice. The load spectrum was a flight-by-flight random-load history including ground-air-ground loads.

500. Reed, D. L.: *Design and Fabrication of Composite Wing Panels Containing a Production Splice*. NASA CR-2549, Aug. 1975.

75N32503#

Bolted specimens representative of both upper and lower wing surface splices of a transport aircraft were designed and manufactured for static and random load tension and compression fatigue testing including ground-air-ground load reversals. The specimens were fabricated with graphite-epoxy composite material. Multiple tests were conducted at various load levels and the results were used as input to a statistical wearout model. The statically designed specimens performed very well under highly magnified fatigue loadings.

501. Rezaizadeh, M. A.; and Mall, S.: *Mixed-Mode Cyclic Debonding of Adhesively Bonded Composite Joints*. NASA CR-177991, Oct. 1985.

86N12255#

A combined experimental-analytical investigation was conducted to characterize the cyclic failure mechanism of a simple composite-to-composite bonded joint. The cyclic debond growth rates were measured.

502. Sawyer, J. W.: *Effect of Preforming Adherends on Static and Fatigue Strength of Bonded Composite Single-Lap Joints*. NASA TP-2324, June 1984.

84N26053#

An analytical and experimental investigation was conducted on bonded composite single-lap joints with the adherends preformed to reduce the angle between the line of action of the applied in-plane force and the bondline. A classical closed-form solution was used to analyze the composite joints with various preform angles and overlap lengths. The adherends of the test specimens were preformed before bonding and during the layup and curing process. Static tests were conducted for preform angles of 0, 5, 10, and 15 deg and overlap lengths of 0.75, 1.75, 2.75, and 3.75 in. A limited fatigue study was conducted for specimens with a 2.75-in. overlap and a preform angle of 5 deg. Results of the analysis showed that preforming the adherends of bonded composite single-lap joints significantly reduced the shear and peel stress concentrations in the adhesive. Experimental results showed that preforming the adherends significantly increased their static and fatigue strength and thus increased the load level for which bonded composite single-lap joints can be designed.

503. Sawyer, J. W.: Effect of Stitching on the Strength of Bonded Composite Single Lap Joints. *24th AIAA Structures, Structural Dynamics and Materials Conference* (Lake Tahoe, Nev., May 2–4), *Collection of Technical Papers, Part 1*, 1983, pp. 510–515. (Available as AIAA Paper 83-0969.)

83A29779#

An experimental investigation has been conducted to determine the effect of stitching on the static and fatigue failure load of bonded composite single lap joints. The variables considered in the static tests included adherend thickness, overlap length, stitch spacing, and number of rows of stitches. A limited fatigue program was conducted for one configuration to compare the fatigue life of stitched and unstitched joints. Up to a 38 percent improvement in static failure load and an order of magnitude increase in fatigue life compared with unstiffened results are obtained by a single row of stitches near the end of the overlap. Additional rows of stitching or different stitch spacing has little effect on static joint failure load. Thicker adherends and larger overlap length result in larger improvements in static failure load with stitching.

504. Shivakumar, K. N.; and Crews, J. H., Jr.: Bolt Clampup Relaxation in a Graphite/Epoxy Laminate. *Long-Term Behavior of Composites* (Proceedings of the Symposium, Williamsburg, Va., Mar. 9–10, 1982), ASTM, 1983, pp. 5–22.

84A24877

A simple bolted joint was analyzed to calculate bolt clampup relaxation for a graphite/epoxy (T300/5208) laminate. A viscoelastic finite-element analysis of a double-lap joint with a steel bolt was conducted. Clampup forces were calculated for various steady-state temperature-moisture conditions using a 20-year exposure duration. The finite element analysis predicted that clampup forces relax even for the room-temperature-dry condition. The relaxations were 8, 13, 20, and 30 percent for exposure durations of 1 day, 1 month, 1 year, and 20 years, respectively. As expected, higher temperatures and moisture levels each increased the relaxation rate. The combined viscoelastic effects of steady-state temperature and moisture appeared to be additive. From the finite-element analysis, a simple equation was used to calculate clampup forces for the same temperature-moisture conditions as used in the finite-element analysis. The two sets of calculated results agreed well.

505. Shivakumar, K. N.; and Crews, J. H., Jr.: *An Equation for Bolt Clampup Relaxation in Transient Environments.* NASA TM-84480, Apr. 1982.

82N25327#

An equation for bolt clampup relaxation for transient temperature-moisture (T-M) conditions was derived starting with a relaxation equation for steady state conditions and then using an incremental time approach that exploits the superposition principle for linear viscoelasticity. The resulting equation uses the initial T-M condition (at the time of clamping), the T-M history after clamping, and elastic clampup coefficients for temperature and moisture changes. The clampup equation was used to calculate the changes in clampup occurring in a T300/5208 graphite/epoxy joint exposed to a one-year history of temperature and moisture. Two cases were considered: one was a dry joint exposed to a relatively humid environment and the other was a nearly saturated joint exposed to an arid environment.

506. Shyprykevich, P.: *Design and Fabrication of Realistic Adhesively Bonded Joints.* NASA CR-166099, May 1983.

83N24876#

Eighteen bonded joint test specimens representing three different designs of a composite wing chordwise bounded splice were designed and fabricated using current aircraft industry practices. Three types of joints (full wing laminate penetration, two side stepped; midthickness penetration, one side stepped; and partial penetration, scarfed) were analyzed using a state-of-the-art elastic joint analysis modified for plastic behavior of the adhesive.

507. Wichorek, G. R.: *Experimental Data on Single-Bolt Joints in Quasi-Isotropic Graphite/Polyimide Laminates.* NASA TP-2015, May 1982.

82N25325#

Sixteen ply, quasi-isotropic laminates of Celanese Celion 6000/PMR-15 and Celion 6000/LARC-160 with a fiber orientation of $(0/45/90/-45)$ were evaluated. Tensile and open hole specimens were tested at room temperature to establish laminate tensile strength and net tensile strength at an unloaded bolt hole. Double lap joint specimens with a single 4.83-mm (0.19 in.) diameter bolt torqued to 1.7 N-m (15 lbf-in.) were tested in tension at temperatures of 116 K ($-250°$F), 297 K ($75°$F), and 589 K ($600°$F). The joint ratios of w/d (specimen width to hole diameter) and e/d (edge distance to hole diameter) were varied from 4 to 6 and from 2 to 4, respectively. The

effects of joint geometry and temperature on failure mode and joint stresses are shown.

508. Wichorek, G. R.; and Crews, J. H., Jr.: *Strength of Graphite/Epoxy Bolted Wing-Skin Splice Specimens Subjected to Outdoor Exposure Under Constant Load and Yearly Fatigue Loading.* NASA TP-2542, 1986.

86N20508#

The results of an experimental study to provide long-term durability data on detailed full-scale graphite/epoxy wing-skin joint designs under environmental exposure and cyclic loading associated with commercial transport aircraft are reported. The specimens consisted of a single-row bolt configuration fabricated from T300/5208 and a double-row bolt configuration fabricated from T300/5209. The unpainted specimens were exposed to the outdoor environment under a sustained tensile load, and at yearly intervals, they were subjected to fatigue loading. Experimental results showed a slight reduction in residual tensile strength for both graphite/epoxy joints under the exposure times and fatigue loadings reported.

509. Wilson, D. W.; Pipes, R. B.; Reigner, D.; and Webster, J.: Mechanical Characterization of PMR-15 Graphite/Polyimide Bolted Joints. *Test Methods and Design Allowables for Fibrous Composites* (Proceedings of the Symposium, Dearborn, Mich., Oct. 2–3, 1979), ASTM, 1981, pp. 195–207.,

81A47812

Data are presented for the static and viscoelastic performance of Celion 6000/PMR-15 graphite/polyimide composite bolted joints over the 21 to 315°C temperature range.

510. Wilson, D. W.; and Pipes, R. B.: *Behavior of Composite Bolted Joints at Elevated Temperature.* NASA CR-159137, Sept. 1979.

79N32574#

Experimental results from an investigation which examines the combined effects of temperature, joint geometry, and out-of-plane constraint upon the response of mechanically fastened composite joints are presented. Data are presented for simulated mechanically fastened joint conditions in two laminate configurations fabricated from Hercules AS/3501-6 graphite-epoxy. Strength and failure mode results are presented for the test temperatures of 21°C, 121°C, and 177°C and for a range of the geometric parameters W/D and e/D from 3.71 to 7.43 and 1.85 to 3.69, respectively. A hole diameter D of 5.16 mm was utilized for all tests.

511. Wilson, D. W.; Gillespie, J. W.; York, J. L.; and Pipes, R. B.: *Failure Analyses of Composite Bolted Joints.* NASA CR-163732, July 1980.

81N11413#

The complex failure behavior exhibited by bolted joints of graphite/epoxy (Hercules AS/3501) was investigated for the net tension, bearing, and shearout failure modes using combined analytical and experimental techniques. Plane stress, linear elastic, finite element methods were employed to determine the two dimensional state of stress resulting from a loaded hole in a finite width, semi-infinite strip. The stresses predicted by the finite element method were verified by experiment to lend credence to the analysis.

Impact dynamics and acoustics.

512. Carden, H. D.: *Impact Dynamics Research on Composite Transport Structures.* NASA TM-86391, Mar. 1985.

85N30367#

The experimental and analytical efforts being undertaken to investigate the response of composite and aluminum structures under crash loading conditions were reviewed. A Boeing 720 airplane was used in the controlled-impact demonstration test. Energy absorption of composite materials, the tearing of fuselage skin panels, the friction and abrasion behavior of composite skins, and the crushing behavior and dynamic response of composite beams were among the topics addressed.

513. Farley, G. L.: *Energy Absorption of Composite Materials.* NASA TM-84638, Mar. 1983.

83N19816#

Results of a study on the energy absorption characteristics of selected composite material systems are presented and the results compared with aluminum. Composite compression tube specimens were fabricated with both tape and woven fabric prepreg using graphite/epoxy, Kevlar/epoxy, and glass/epoxy. Chamfering and notching one end of the composite tube specimen reduced the peak load at initial failure without altering the sustained crushing load and prevented catastrophic failure. Static compression and vertical impact tests were performed on 128 tubes.

514. Goodall, R. G.; and Painter, G. W.: *Conceptual Design Study of Advanced Acoustic Composite Nacelle.* NASA CR-132649, May 1975.

75N23568#

Conceptual nacelle designs for wide-bodied and for advanced-technology transports were studied with the objective of achieving significant reductions in community noise with minimum penalties in airplane weight, cost, and operating expense by the application of advanced composite materials to nacelle structure and sound suppression elements. Nacelle concepts using advanced liners, annular splitters, radial splitters, translating centerbody inlets, and mixed-flow nozzles were evaluated and a preferred concept selected. A preliminary design study of the selected concept, a mixed flow nacelle with extended inlet and no splitters, was conducted and the effects on noise, direct operating cost, and return on investment determined.

515. Grosveld, F. W.; and Metcalf, V. L.: Modal Response and Noise Transmission of Composite Panels. *26th AIAA Structures, Structural Dynamics and Materials Conference* (Orlando, Fla., Apr. 15–17), *Technical Papers, Part 2*, 1985, pp. 617–627. (Available as AIAA Paper 85-0789.)

85A30391#

Noise transmission through flat, angular, fiber-reinforced composite panels was investigated experimentally and analytically. A modal decomposition technique was used to obtain solutions to the governing differential equation of motion. Experimental modal analysis was performed in order to confirm the theoretical results. The test specimens were cross-ply and angle-ply composite panels made of various concentrations of fiberglass, graphite, or aramid fibers embedded in epoxy resin. The experimental results showed good agreement with the theoretical calculations. Graphs of the transmission loss characteristics of the different composite panels are provided.

516. Jackson, K. E.: *Abrasion Behavior of Aluminum and Composite Skin Coupons, Stiffened Skins, and Stiffened Panels Representative of Transport Airplane Structures.* NASA TP-2520, Nov. 1985.

86N13316#

A three-phase investigation was conducted to compare the friction and wear response of aluminum and graphite-epoxy composite materials when subjected to loading conditions similar to those experienced by the skin panels on the underside of a transport airplane during an emergency belly landing on a runway surface. The first phase involved a laboratory test which used a standard belt sander to provide the sliding abrasive surface. Small skin-coupon test specimens were abraded over a range of pressures and

velocities to determine the effects of these variables on the coefficient of friction and wear rate. The second phase involved abrading I-beam stiffened skins on an actual runway surface over the same range of pressures and velocities used in the first phase. In the third phase, large stiffened panels which most closely resembled transport fuselage skin construction were abraded on a runway surface. This report presents results from each phase of the investigation and shows comparisons between the friction and wear behavior of the aluminum and graphite-epoxy composite materials.

517. Jackson, K. E.: *Friction and Wear Behavior of Aluminum and Composite Airplane Skins.* NASA TP-2262, Feb. 1984.

84N17175#

Friction and wear behavior was determined for small skin specimens under abrasive loading conditions typical of those occurring on the underside of a transport airplane during emergency belly landing.

518. Koval, L. R.: Sound Transmission Into a Laminated Composite Cylindrical Shell. *Journal of Sound and Vibration*, vol. 71, Aug. 22, 1980, pp. 523–530.

80A52724

A mathematical model is presented for the transmission of an oblique plane sound wave into a laminated composite circular cylindrical shell. Numerical results are obtained for geometry typical of a narrow-body jet transport. Results indicate that from the viewpoint of noise attenuation a laminated composite shell does not appear to offer any significant advantage over an aluminum shell. However, the transmission loss of a laminated composite shell is sensitive to the orientation of the fibers and this suggests the possibility of using a laminated composite shell to tailor the noise attenuation characteristics to meet a specific need.

519. Koval, L. R.: *Field-Incidence Transmission of Treated Orthotropic and Laminated Composite Panels.* NASA TM-85680, Aug. 1983.

84N10911#

An analytical model was developed for the field-incidence transmission loss of an orthotropic or laminated composite infinite panel with layers of various noise insulation treatments. The model allows for four types of treatments, impervious limp septa, orthotropic trim panels, porous blankets, and air spaces, while it also takes into account the effects

of forward speed. Agreement between the model and transmission loss data for treated panels is seen to be fairly good overall. In comparison with transmission loss data for untreated composite panels, excellent agreement occurred.

520. Lameris, J.; Stevenson, S.; and Streeter, B.: *Study of Noise Reduction Characteristics of Composite Fiber-Reinforced Panels.* NASA CR-168745, Mar. 1982.

82N21999#

A composite panel program was initiated to study the effects of some of the parameters that affect noise reduction of these panels. The fiber materials and ply orientation were chosen to be variables in the test program. It was found that increasing the damping characteristics of a structural panel will reduce the vibration amplitudes at resonant frequencies with attendant reductions in sound. Test results for a dynamic absorber, a tuned damper, are presented and evaluated.

521. McGary, M. C.: *Separation of Airborne and Structureborne Noise Radiated by Plates Constructed of Conventional and Composite Materials With Applications for Prediction of Interior Noise Paths in Propeller Driven Aircraft.* NASA TM-87414, Jan. 1986.

86N18121#

A new diagnostic method which permits the separation and prediction of the fully coherent airborne and structureborne components of the sound radiated by plates or thin shells has been developed. Analytical and experimental studies of the proposed method were performed on plates constructed of both conventional and composite materials. The results of the study indicate that the proposed method can be applied to a variety of aircraft materials, could be used in flight, and has fewer encumbrances than the other diagnostic tools currently available.

522. Mei, C.; and Prasad C. B.: *Nonlinear Random Response of Composite Panels.* NASA CR-176689, Sept. 1985.

86X10236#

Computer programs were developed for the prediction of large amplitude random responses in isotropic and laminated composite rectangular thin plates. The mathematical formulations of the programs are reported. The programs have been used for determining root-mean-square (RMS) strains for test panels at various acoustic excitations.

523. Metcalf, V. L.; and Grosveld, F. W.: Noise Transmission Characteristics of Aircraft-Type Composite Panels. SAE Paper 850878, 1985.

85A50110

The noise transmission loss characteristics of 28 flat, fiber reinforced, composite panels and 4 aluminum panels, with dimensions of 24.1 cm by 40.6 cm, have been experimentally investigated in the NASA Langley Research Center noise transmission loss apparatus. To support the acoustic data, modal frequencies and modal damping for several mode shapes of each of the test panels were extracted experimentally. The transmission loss of composite panels is compared with that of aluminum panels of equal critical shear load.

524. Navaneethan, R.; Quayle, B.; Stevenson, S.; and Graham, M.: *Study of Noise Reduction Characteristics of Double-Wall Panels.* NASA CR-170308, May 1983.

83N25497#

The noise reduction characteristics of general aviation type, flat, double-wall structures were investigated. The experimental study was carried out on 20-by-20-inch panels with an exposed area of 18 by 18 inches. A frequency range from 20 to 5000 Hz was covered. The experimental results, in general, follow the expected trends.

525. Navaneethan, R.: *Study of Noise Reduction Characteristics of Multilayered Panels and Dual-Pane Windows With Helmholtz Resonators.* NASA CR-164370, May 1981.

81N24857#

The experimental noise attenuation characteristics of flat, general aviation type, multilayered panels are discussed. Experimental results of stiffened panels, damping tape, honeycomb materials, and sound absorption materials are presented. Single degree-of-freedom theoretical models were developed for sandwich type panels with both shear resistant and non-shear resistant core material. The concept of Helmholtz resonators used in conjunction with dual-pane windows in increasing the noise reduction over a small range of frequency was tested. It is concluded that the stiffening of the panels increases the low frequency noise reduction.

526. Norton, H. T., Jr.: *Summary of Recent Design Studies of Advanced Acoustic-Composite Nacelles.* NASA TM X-72774, Sept. 1979.

75N32094#

The results are summarized from recent NASA-sponsored studies of advanced acoustic-composite nacelles. These studies were conducted with the objective of achieving significant reductions in community noise and/or fuel consumption with minimum penalties in airplane weights, cost, and operating expense. The results indicate that the use of composite materials offers significant potential weight and cost savings and results in reduced fuel consumption and noise when applied to nacelles. The most promising concept for realizing all of these benefits was a long duct, mixed flow acoustic-composite nacelle with advanced acoustic liners.

527. Patrick, H. V. L.: *Investigation of Composite Materials Property Requirements for Sonic Fatigue Research.* NASA TM-87601, Aug. 1985.

86N10913#

Experimental techniques for determining the extensional and bending stiffness characteristics for symmetric laminates are presented. Vibrational test techniques for determining the dynamic modulus and material damping are also discussed. Partial extensional stiffness results initially indicate that the laminate theory used for predicting stiffness is accurate. It is clearly shown that the laminate theory can only be as accurate as the physical characteristics describing the lamina, which may vary significantly. It is recommended that all of the stiffness characteristics in both extension and bending be experimentally determined to fully verify the laminate theory. Dynamic modulus should be experimentally evaluated to determine if static data adequately predicts dynamic behavior. Material damping should also be ascertained because laminate damping is an order of magnitude greater than found in common metals and can significantly effect the displacement response of composite panels.

528. Rennison, D. C.; Wilby, J. F.; and March, A. H.: *Interior Noise Control Prediction Study for High-Speed Propeller-Driven Aircraft.* NASA CR-159200, Sept. 1979.

80N25102#

An analytical model was developed to predict the noise levels inside propeller-driven aircraft during cruise at $M = 0.8$. The model was applied to three study aircraft with fuselages of different sizes (wide body, narrow body, and small diameter) in order to determine the noise reductions required to achieve the goal of an A-weighted sound level which does not exceed 80 dB. The model was then used to determine noise control methods which could achieve the

required noise reductions. Two classes of noise control treatments were investigated: add-on treatments which can be added to existing structures, and advanced concepts which would require changes to the fuselage primary structure. Only one treatment, a double wall with limp panel, provided the required noise reductions. Weight penalties associated with the treatment were estimated for the three study aircraft.

529. Revell, J. D.; Balena, F. J.; and Koval, L. R.: *Analytical Study of Interior Noise Control by Fuselage Design Techniques on High-Speed, Propeller-Driven Aircraft.* NASA CR-159222, Apr. 1980.

80N25105#

The acoustical treatment mass penalties required to achieve an interior noise level of 80 dBA for high speed, fuel efficient, propfan-powered aircraft are determined. The prediction method used is based on theory developed for the outer shell dynamics and a modified approach for add-on noise control element performance. The present synthesis of these methods is supported by experimental data. Three different sized aircraft are studied, including a widebody, a narrowbody, and a business sized aircraft.

530. Roussos, L. A.; Powell, C. A.; Grosveld, F. W.; and Koval, L. R.: Noise Transmission Characteristics of Advanced Composite Structural Materials. AIAA Paper 83-0694, 1983.

83A25915#

Theoretical and experimental results from a study of noise transmission properties of large unstiffened panels which simulated aircraft outer skins and interior trim are reported. The investigation was performed to define the effects of composite structures on fuselage noise transmission relative to the transmissivity of aluminum structures. One-third octave band measurements were obtained in a two-room facility for measuring transmission loss. Center frequencies of at least 100 Hz were used, and 14 different composite panels, including samples of Kevlar, fiberglass, and graphite, were examined.

531. Vaicaitis, R.; and Bofilios, D. A.: Response of Double Wall Composite Shells. *26th AIAA Structures, Structural Dynamics and Materials Conference* (Orlando, Fla., Apr. 15–17), *Technical Papers, Part 2,* 1985, pp. 110–119. (Available as AIAA Paper 85-0604.)

85A30332#

This paper presents an analytical study of double wall laminate cylindrical shell response to random loads. A soft viscoelastic core with dilatational modes included is used. The theory of laminate shells is simplified by assumptions similar to those in the Donnell-Mushtari development for isotropic shells. Modal solutions of simply supported shells are obtained. Modal frequencies and deflection response spectral densities are determined. It is found that this approach allows for easy parametric evaluation and that by proper selection of dynamic parameters, viscoelastic core characteristics, and fiber reinforcement orientation, vibration response can be reduced.

532. Vaicaitis, R.; and Bofilios, D. A.: Noise Transmission of Double Wall Composite Shells. *Fluid-Structure Interaction and Aerodynamics Damping* (Proceedings of the Tenth Biennial Conference on Mechanical Vibration and Noise, Cincinnati, Ohio, Sept. 10–13), ASME, 1985, pp. 51–64.

86A26890#

This paper describes analytical studies of noise transmission of double wall laminated composite cylindrical shells of finite extent. The theoretical solutions of the governing acoustic-structural equations are obtained utilizing modal decomposition. Results indicate that, from the viewpoint of noise attenuation, the composite shell does not seem to offer advantage over an aluminum shell. However, the orientation of the fibers of the material characteristics of the soft viscoelastic core can be tailored to meet the specific needs of noise attenuation.

Graphite-polyimide and polymer development

533. Awerbuch, J.; Perkinson, H. E.; and Kamel, I. L.: Notched Graphite/Polyimide Composites at Room and Elevated Temperatures—Nondestructive Test Techniques. NASA CR-159375, Aug. 1980.

81N20185#

The fracture behavior in graphite/polyimide (Gr/PI) Celion 6000/PMR-15 composites was characterized. Emphasis was placed on the correlation between the observed failure modes and the deformation characteristics of center-notched Gr/PI laminates. Crack tip damage growth, fracture strength and notch sensitivity, and the associated characterization methods were also examined.

534. Baucom, R. M.; and Kidder, P. W.: Effect of Storing B-Staged Graphite/Polyimide on Fabrication of Structural Elements. *Material and Process Applications—Land, Sea, Air, Space* (Proceedings of

the Twenty-Sixth National Symposium and Exhibition, Los Angeles, Calif., Apr. 28–39), Society for the Advancement of Material and Process Engineering, 1981, pp. 89–100.

81A44332

Graphite/polyimide structural elements have been successfully fabricated by forming and curing B-staged Celion 6000/PMR-15 and Celion 6000/LaRC-160 laminates in heated matched metal tooling. The B-staged laminates were subjected to storage for various periods of time up to six months at ambient shop conditions and high humidity prior to fabrication into structural elements to determine the processability of the graphite/polyimide after exposure to these conditions. Changes in processability were monitored by measuring flexure and interlaminar shear strengths, glass transition temperature, moisture content, and formability of the B-staged laminates.

535. Bergren, O. D.; Lockerby, S. C.; and Maximovich, M. G.: Fabrication of LaRC 160/NR150B2 Hybrid Matrix Polyimide Composites. *Materials 1980* (Proceedings of the Twelfth National Technical Conference, Seattle, Wash., Oct. 7–9), Society for the Advancement of Material and Process Engineering, 1980, pp. 31–53.

81A43604

Graphite-polyimide composite materials exhibit high performance, low weight, and good thermal-oxidative stability, making them promising candidates for Space Shuttle applications. However, no single material is excellent in all respects: processability, thermal-oxidative stability, toughness, and mechanical properties. This paper describes the development of hybrid matrix composite laminates that combine the attributes of two polyimide resins, NR150B2 and LaRC 160, while avoiding their drawbacks. High quality laminates were fabricated from each resin system, and hybrid laminates were successfully co-cured, evaluated, and optimized.

536. Butler, J. M.; and St. Clair, T. L.: *Tackifier for Addition Polyimides.* NASA TM-81883, Oct. 1980.

80N33570#

A modification to the addition polyimide, LaRC-160, was prepared to improve tack and drape and increase prepreg out-time. The essentially solventless, high viscosity laminating resin is synthesized from low-cost liquid monomers. The modified version takes advantage of a reactive, liquid plasticizer which is used in place of solvent and helps solve a

major problem of maintaining good prepreg tack and drape.

537. Chase, V. A.; and Harrison, E. S.: *Develop and Demonstrate Manufacturing Processes for Fabricating Graphite Filament Reinforced Polyimide (Gr/PI) Composite Structural Elements.* NASA CR-172497, Feb. 1985.

85N20019#

A study was conducted to assess the merits of using graphite/polyimide, NR-150B2 resin, for structural applications on advanced space launch vehicles.

538. Dara, P. H.; and Loos, A. C.: *Thermoplastic Matrix Composite Processing Model.* NASA CR-176639, Sept. 1985.

86N21619#

The effects the processing parameters of pressure, temperature, and time have on the quality of continuous graphite fiber reinforced thermoplastic matrix composites were quantitatively assessed by defining the extent to which intimate contact and bond formation has occurred at successive ply interfaces. Two models are presented predicting the extents to which the ply interfaces have achieved intimate contact and cohesive strength.

539. Davis, J. G., Jr.: *Composites for Advanced Space Transportation Systems (CASTS).* NASA TM-80038, Mar. 1979.

79N20192#

A summary is given of the in-house and contract work accomplished under the CASTS Project. In July 1975 the CASTS Project was initiated to develop graphite fiber/polyimide matrix (GR/PI) composite structures with 589 K (600°F) operational capability for application to aerospace vehicles. Major tasks include: (1) screening composites and adhesives, (2) developing fabrication procedures and specifications, (3) developing design allowables test methods and data, and (4) design and test of structural elements and construction of an aft body flap for the Space Shuttle Orbiter Vehicle which will be ground tested. Portions of the information are from ongoing research and must be considered preliminary. The CASTS Project was scheduled to be completed in September 1983.

540. Delano, C. B.; and Kiskiras, C. J.: *Development of an Impact- and Solvent-Resistant Thermoplastic Composite Matrix.* NASA CR-172568, June 1985.

85N31239#

The polyimide from BTDA 1,6-hexanediamine and m-phenylenediamine was selected from a prior study for the present study. Methods to prepare prepreg which would provide low-void composites at low molding pressures from the thermoplastic polyimide were studied. Cresol solutions of the polyimide were applied to a balanced weave carbon fabric.

541. Dexter, H. B.; and Davis, J. G., Jr., eds.: *Graphite/Polyimide Composites—Conference on Composites for Advanced Space Transportation Systems.* NASA CP-2079, Aug. 1979.

79N30297#

Technology developed under the Composites for Advanced Space Transportation System Project is reported. Specific topics covered include fabrication, adhesives, test methods, structural integrity, design and analysis, advanced technology developments, high temperature polymer research, and the state of the art of graphite/polyimide composites.

542. Edelman, R.: *Study of Catalyst Cured LaRC-160/Celion 6000.* NASA CR-159335, Sept. 1980.

80N31480#

LaRC-160 polyimide was modified with a high temperature peroxide catalyst, USP-138, in an attempt to reduce the final cure temperature to below 316°C. This effort was directed at obtaining a material for 177°C use that would cure at similar temperatures to a high performance epoxy while still maintaining the good moisture resistance of a polyimide material.

543. Freeman, W. T., Jr.; and Baucom, R. M.: Graphite Polyimide Fabrication Research for Supersonic Cruise Aircraft. *The 1980's—Payoff Decade for Advanced Materials* (Proceedings of the Twenty-Fifth National Symposium and Exhibition, San Diego, Calif., May 6-8), Society for the Advancement of Material and Process Engineering, 1980, pp. 418–432.

81A22665

Advanced fabrication processes and adhesive bonding methods have been developed for the fabrication of full-scale fiberglass-polyimide honeycomb stiffened HTS-1 graphite/PMR-15 polyimide panels that meet the design criteria for an upper wing panel of the NASA YF-12 aircraft. Detailed manufacturing, bonding, and autoclave cure procedures are presented.

544. Frost, R. K.; and Wykes, D. H.: Fabrication of Bonded Graphite/Polyimide Structures for Advanced Aerospace Applications. *Material and Process Advances '82* (Proceedings of the Fourteenth National SAMPE Technical Conference, Atlanta, Ga., Oct. 12–14), Society for the Advancement of Material and Process Engineering, 1982, pp. 418–425.

83A23636

Tooling concepts and manufacturing techniques are examined as part of the structural development of graphite/polyimide composite material systems for the Space Shuttle Orbiter and other space transportation systems.

545. Frost, R. K.; Jones, J. S.; and Dynes, P. J.: *Development and Demonstration of Manufacturing Processes for Fabricating Graphite/LARC 160 Polyimide Structural Elements.* NASA CR-165809, Dec. 1981.

82N22315#

The development and demonstration of manufacturing technologies for the structural application of Celion graphite/LARC 160 polyimide composite material are discussed. Process development and fabrication of demonstration components are discussed.

546. Hergenrother, P. M.; and Johnston, N. J.: Status of High-Temperature Laminating Resins and Adhesives. *Resins for Aerospace* (Proceedings of the Symposium, Honolulu, Hawaii, Apr. 3–6, 1979), American Chemical Society, 1980, pp. 3–13.

81A20852

High-temperature polymers now being developed as adhesives and composite matrices are reviewed, including aromatic polyimides, polybenzimidazoles, polyphenylquinoxalines, nadic end-capped imide oligomers, maleimide end-capped oligomers, and acetylene-terminated imide oligomers. The mechanical properties of laminates based on these resins are reported together with preliminary test results on the adhesive properties for titanium-to-titanium and composite-to-composite lap shear specimens.

547. Hergenrother, P. M.: Adhesive and Composite Evaluation of Acetylene-Terminated Phenylquinoxaline Resins. *Polymer Engineering and Science,* vol. 21, no. 16, Nov. 1981, pp. 1072–1078.

82A15875

A series of acetylene-terminated phenylquinoxaline (ATPQ) oligomers of various molecular weights were prepared and subsequently chain extended by the thermally induced reaction of the ethynyl groups. The processability and thermal properties of these oligomers and their cured resins were compared with that of a relatively high molecular weight linear polyphenylquinoxaline (PPQ) with the same chemical backbone. The ATPQ oligomers exhibited significantly better processability than the linear PPQ but the PPQ displayed substantially better oxidation stability. Adhesive (Ti/Ti) and composite (graphite filament reinforcement) work was performed to evaluate the potential of these materials for structural applications.

548. Hergenrother, P. M.; Harrison, E. S.; and Gosnell, R. B.: New Cyanate and Cyanamide Resins as Composite Matrices. *Selective Application of Materials for Products and Energy* (Proceedings of the Twenty-Third National Symposium and Exhibition, Anaheim, Calif., May 2–4), Society for the Advancement of Material and Process Engineering, 1978, pp. 506–519.

79A20832

The synthesis, polymerization, and laminate properties of two dicyanates, copolymers thereof, and a dicyanamide were studied. The effect of humidity aging (322 K (49°C), 95% relative humidity) and also aging at 505 K (232°C) in circulating air on the RT and elevated temperature properties of HT-S and T-300 unidirectional laminates was determined.

549. Hoggatt, J. T.; Hunter, A. B.; and Sheppard, C. H.: Reproducibility of PMR-15 Graphite Polyimide Material. *Proceedings of the 33rd Reinforced Plastics/Composites Institute Annual Conference* (Washington, D.C., Feb. 7–10), Society of the Plastics Industry, Inc., 1978, pp. 15-E1–15-E4.

79A15536#

Investigation was made into the reproducibility of PMR polyimide/graphite prepreg reinforcement and the influence of these variabilities on processing characteristics and final composite properties. The purpose of these investigations was to establish specification controls and limits of acceptance/rejection on incoming materials. Prepreg samples were chemically analyzed and the results compared to the processing and property results of the resulting composites. The findings from these comparisons are discussed.

550. Kray, R. J.; and Ferrara, A.: Fabrication and Evaluation of RX-6450/NCNS-13P/Composites.

Materials 1980 (Proceedings of the Twelfth National Technical Conference, Seattle, Wash., Oct. 7–9), Society for the Advancement of Material and Process Engineering, 1980, pp. 505–516.

81A43634#

The chemistry, processing, and laminate properties of NCNS-13P resin are described. The effects on graphite fiber laminates of temperature, oxidative aging, water absorption, and humidity-temperature aging were determined. Glass laminates were evaluated for fire resistance and smoke and toxic gas evolution under flaming conditions.

551. Maximovich, M.; Wereta, A.; and Galeos, R.: *Chemical and Rheological Evaluation of Polyimides.* NASA CR-165817, May 1982.

82N23320

Chemical and rheological evaluation techniques were developed for polyimide composite matrix resins. LARC 160 (AP-22), LARC 160 (Curithane 103), PMR 15, and PISO2 resins were investigated. Liquid and solid state rheological techniques were used to develop processing parameters, and woven graphite reinforced laminates were successfully fabricated using these processing parameters.

552. Maximovich, M. G.; Bergren, O.; and Lockerby, S.: *Development of LaRC 160/NR150B2 Polyimide Graphite Hybrid Composites.* NASA CR-159196, Mar. 1980.

80N20315#

A method for co-curing NR150B2 and LaRC 160 prepregs into hybrid composites was developed. The processing characteristics and the properties of the hybrid composites were compared with those of laminates fabricated from the individual component prepregs. Resin forms were selected and optimized and a new NR150 formulation was investigated. Quality control techniques were evaluated and developed, high quality laminates were fabricated from both individual resin systems, and hybrid laminates were successfully co-cured. Optimum hybrid forms were investigated and several novel approaches were explored. An optimum hybrid system was developed that utilizes an LaRC curing schedule but shows no degradation of mechanical properties after aging 500 hr in air at 260°C.

553. Nelson, J. B.: Thermal Aging of Graphite/Polyimide Composites. *Long-Term Behavior of Composites* (Proceedings of the Symposium, Williamsburg, Va., Mar. 9–10, 1982), ASTM, 1983, pp. 206–221.

84A24888

Celion 6000/PMR-15, Celion 6000/LARC-160, and Celion 6000/RK-99 graphite/polyimides were aged in circulating air ovens at temperatures of 202°, 232°, 260°, and 288°C for various times, up to 15,000 hours. The relative thermal-oxidative stability from highest to lowest is as follows: PMR-15, LARC-160, RK-99, Celion 6000/PMR-15, and Celion 6000/LARC-160 laminates retained at least 80 percent of their initial flexural strengths for the duration of aging at each temperature. Celion 6000/RK-99 laminates exhibited a 20 to 50 percent loss of flexural strength at all aging temperatures. All three graphite/polyimide laminate materials degraded preferentially at the specimen edge perpendicular to the fibers.

554. Nelson, J. B.: *Long-Term Thermal Aging of Two Graphite-Polyimide Composite Materials.* NASA TP-2369, 1984.

85N12094#

Two graphite-polyimide composite materials were aged in circulating air ovens at temperatures of 204°C, 232°C, 260°C, and 288°C for various times up to 25000 hours. The composites were (1) Celanese Celion 6000 graphite fiber and PMR-15 polyimide resin (Celion/PMR-15) and (2) Celion 6000 graphite fiber and LARC-160 polyimide resin (Celion/LARC-160). Three unidirectional specimen geometries were studied: short beam shear (SBS) specimens, flexure specimens, and 153 mm square panels. The interior regions of the square panels exhibited only minor property degradation. The individually aged SBS and flexure specimens exhibited large reductions in strengths after aging. Both laminate materials cracked and degraded preferentially at the specimen edge perpendicular to the fibers.

555. Nelson, J. B.: *Long-Term Thermal Aging of Celion/V378A Composite Materials.* NASA TM-86362, June 1985.

85N31236#

Celion 6000/V378A graphite/bismaleimide composite materials were aged in air at temperatures of 177°, 204°, 232°, and 260°C for various times up to 15,000 hours. Three unidirectional specimen types were aged: short beam shear (SBS), flexure, and 153 mm square panels. Aged specimens of V378A laminates exhibited excellent thermal stability. Extensive cracking was observed during aging on the 0 deg edges of the unidirectional laminates. These cracks penetrated as deep as 12 mm from the edge. The cracking appeared to have little or no effect on

the observed properties of the laminates. The study indicates that the useful life of unrestrained unidirectional graphite/V378A laminates is 10,000 hours or greater at 177°C to 232°C and 2,000 hours at 260°C.

556. Rosen, B. W.; Nagarkar, A. P.; and Hashin, Z.: *Thermomechanical Response of Gr/Pi Composites.* NASA CR-165753, Mar. 1983.

82N15124#

The effects of temperature changes upon the stresses and strains in composite laminates having carbon fibers in a polyimide matrix were evaluated. Composites having laminae in which there were nonlinear stress-strain relations for stresses transverse to the fibers and for axial shear stresses were treated. Material properties were considered to be temperature dependent. Separately, effects of laminae viscoelastic response were also treated. The results suggest that, for this material, nonlinearities due either to stress or time dependent effects do not appear to be of major practical importance for conventional high temperature composite structures.

557. St. Clair, A. K.; and St. Clair, T. L.: A Multi-Purpose Thermoplastic Polyimide. *SAMPE Quarterly*, vol. 13, Oct. 1981, pp. 20–25.

82A21361

A linear thermoplastic polyimide, LARC-TPI, has been characterized and developed for a variety of high-temperature applications. In its fully imidized form, this new material can be used as an adhesive for bonding metals such as titanium, aluminum, copper, brass, and stainless steel. LARC-TPI is being evaluated as a thermoplastic for bonding large pieces of polyimide film to produce flexible, 100% void-free laminates for flexible circuit applications. The further development of LARC-TPI as a potential molding powder, composite matrix resin, high-temperature film, and fiber is also discussed.

558. St. Clair, T. L.; and Yamaki, D. A.: A Thermoplastic Polyimidesulfone. *Polyimides: Synthesis, Characterization, and Applications, Volume 1,* Plenum Press, 1984, pp. 99–116.

85A21483

A polymer system has been prepared which has the excellent thermoplastic properties generally associated with polysulfones and the solvent resistance and thermal stability of aromatic polyimides. This material, with improved processability over the base polyimide, can be processed in the 260°–325°C range in such a manner as to yield high quality, tough

unfilled moldings; strong, high-temperature-resistant adhesive bonds; and well consolidated, graphite-fiber-reinforced moldings (composites). The unfilled moldings have physical properties that are similar to aromatic polysulfones which demonstrates the potential as an engineering thermoplastic. The adhesive bonds exhibit excellent retention of initial strength levels even after thermal aging for 5000 hours at 232°C. The graphite-fiber-reinforced moldings have mechanical properties which makes this polymer attractive for the fabrication of structural composites.

559. St. Clair, T. L.; and Jewell, R. A.: LaRC-160—A New 550°F Polyimide Laminating Resin. *Bicentennial of Materials* (Proceedings of the Eighth National Technical Conference, Seattle, Wash., Oct. 12–14), Society for the Advancement of Material and Process Engineering, 1976, pp. 82–93.

77A36677

A new addition polyimide, LaRC-160, based on liquid monomers has been prepared. The essentially solventless prepreg retains good drape and tack. Unidirectional HT-S composites were fabricated and tested before and after aging at temperatures between 505 K and 589 K (450°F and 600°F). Excellent retention of mechanical properties was realized after 2000 hours at 547 K (525°F). This material should be a candidate for solventless prepregging due to the liquid nature of the monomeric mixture.

560. St. Clair, T. L.; and Jewell, R. A.: *Solventless LARC-160 Polyimide Matrix Resin.* NASA TM-74994, 1978.

76N16187#

The addition polyimide, LARC-160, which was originally synthesized from low cost liquid monomers as a laminating resin in ethanol, was prepared as a solventless, high viscosity, neat liquid resin. The resin was processed by hot-melt coating techniques into graphite prepreg with excellent tack and drape. Comparable data on graphite reinforced laminates made from solvent-coated and various hot-melt coated prepreg were generated. LARC-160, because of its liquid nature, can be easily autoclave processed to produce low void laminates. Liquid chromatographic fingerprints indicate good reaction control on resin scale ups. Minor changes in monomer ratios were also made to improve the thermal aging performance of graphite laminates.

561. Sheppard, C. H.; Hoggatt, J. T.; and Hunter, A. B.: Development and Demonstration of Manufacturing Processes for Fabricating Graphite/

PMR-15 Polyimide Structures. *New Horizons— Materials and Processes for the Eighties* (Proceedings of the Eleventh National Conference, Boston, Mass., Nov. 13-15), Society for the Advancement of Material and Process Engineering, 1979, pp. 40-48.

80A34755

The work included establishing controls on the polymer, the prepreg, composite fabrication, and quality assurance, as well as fabrication of structural elements to demonstrate the developed materials and processes. The fabricated structures were hat sections, I-beam sections, honeycomb sandwich structures, and molded graphite-reinforced fittings. The graphite/PMR-15 polyimide system was shown to be well suited for use in the 550-600°F temperature range; the processing techniques developed were proved and found potentially useful for other commercially available systems.

Materials and structures for helicopters.

562. Adams, K. M.; and Lucas, J. J.: *Study To Investigate Design, Fabrication and Test of Low Cost Concepts for Large Hybrid Composite Helicopter Fuselage, Phase 2.* NASA CR-145167, Apr. 1977.

77N24097#

The development of a frame/stringer/skin fabrication technique for composite airframe construction was studied as a low cost approach to the manufacture of large helicopter airframe components. A center cabin section of the Sikorsky CH-53D was selected for evaluation as a composite structure. The design, as developed, is composed of a woven Kevlar-49/epoxy skin and graphite/epoxy frames and stringers. Bolted composite channel sections were selected as the optimum joint construction.

563. Adams, K. M.; and Lucas, J. J.: *Study To Investigate Design, Fabrication and Test of Low Cost Concepts for Large Hybrid Composite Helicopter Fuselage, Phase 1.* NASA CR-132731, June 1975.

75N33031#

The development of a frame/stringer/skin fabrication technique for composite airframe construction was studied as a low cost approach to the manufacture of large helicopter airframe components. A center cabin section of the Sikorsky CH-53D helicopter was selected for evaluation as a composite structure. The design, as developed, is composed of a woven Kevlar-49/epoxy skin and graphite/epoxy frames and stringers. To support the selection of this specific design concept a materials study was

conducted to develop and select a cure compatible graphite and Kevlar-49/epoxy resin system, and a foam system capable of maintaining shape and integrity under the processing conditions established. The materials selected were Narmco 5209/Thornel T-300 graphite, Narmco 5209/Kevlar-49 woven fabric, and Stathane 8747 polyurethane foam.

564. Baker, D. J.: *Stress Analysis of Advanced Attack Helicopter Composite Main Rotor Blade Root End Lug.* NASA TM-84578, Dec. 1982.

83N15361#

Stress analysis of the Advanced Attack Helicopter (AAH) composite main rotor blade root end lug is described. The stress concentration factor determined from a finite element analysis is compared to an empirical value used in the lug design. The analysis and test data indicate that the stress concentration is primarily a function of configuration and independent of the range of material properties typical of Kevlar-49/epoxy and glass/epoxy.

565. Bielawa, R. L.; Cheney, M. C., Jr.; and Novak, R. C.: *Investigation of a Bearingless Helicopter Rotor Concept Having a Composite Primary Structure.* NASA CR-2637, Oct. 1976.

77N10044#

Experimental and analytical investigations were conducted to evaluate a bearingless helicopter rotor concept.

566. Bowman, L. M.: *Effect of Measured Material Properties on the Finite Element Analysis of an OH-58 Composite Tail Boom.* NASA TM-86430, Oct. 1985.

86N12259#

A static and dynamic finite element analysis was conducted on a U.S. Army OH-58 composite tail boom and compared with test data. To determine the effect of using measured material properties, static and dynamic finite element analyses were conducted for three fiber-volume conditions of 45, 48, and 50 percent. The static and dynamic model with the 45-percent fiber-volume graphite skins gives the closest agreement with test data.

567. Dexter, H. B.; and Huston, R. J.: *Assessment of Advanced Materials for Rotorcraft.* NASA TM-85675, Aug. 1983.

83X10290#

The application of advanced materials and structural concepts to rotorcraft was assessed. State-of-the-art, potential benefits, problem areas, and technology requirements of advanced composites and metals for civilian and military helicopters are presented.

568. Ezzo, M. B.; and Lowry, D. W.: *Environmental and Structural Assessment of Four Year In-Commercial Service Composite Helicopter Structures.* Proceedings of the *40th American Helicopter Society Annual Forum* (Arlington, Va., May 16–18), 1984, pp. 105–112.

85A31958#

This paper presents the technical background for including environmental effects in the design of helicopter composite structures. This effort was supported by test results of components and panels having up to four years of field exposure. Full-scale static and fatigue tests were conducted on composite components removed from S-76 helicopters used in commercial operations in the Gulf Coast region of Louisiana. Small scale tests were conducted using coupons removed from panels exposed to the outdoor environment in Stratford, Connecticut, and West Palm Beach, Florida. Moisture levels were compared with predicted values. Mechanical test results were compared to initial certification tests for component strength and to baseline laboratory coupon testing.

569. Gleich, D.: *Design Study of Prestressed Rotor Spar Concept.* NASA CR-159086, Jan. 1980.

80N17062#

Studies on the Bell Helicopter 540 Rotor System of the AH-1G helicopter were performed. The stiffness, mass, and geometric configurations of the Bell blade were matched to give a dynamically similar prestressed composite blade. A multi-tube, prestressed composite spar blade configuration was designed for superior ballistic survivability at low life-cycle cost.

570. Hoffstedt, D. J.: *Research Requirements To Reduce Empty Weight of Helicopters by Use of Advanced Materials.* NASA CR-145113, Dec. 1976.

77N18130#

Utilization of the new lightweight, high-strength, aerospace structural-composite (filament/matrix) materials, when specifically designed into a new aircraft, promises reductions in structural empty weight of 12 percent at recurring costs competitive with metals. A structural empty-weight reduction of 12 percent was shown to significantly reduce energy consumption in modern high-performance helicopters.

571. Kay, B. F.; Lowry, D. W.; and Rich, M. J.: *Study To Investigate Design, Fabrication and Test of Low-Cost Concepts for Large Hybrid Composite Helicopter Fuselage, Phase 3.* NASA CR-158988, Feb. 1979.

79X10013#

Design concepts were developed for the two most difficult joint areas in the transmission support area of the airframe. Three coupons of the bolted joint of the transmission frame/beam to the main rotor transmission and the beam/frame (web) intersection were statically tested. The transmission attachment joint analysis correlation with test results was excellent in that the analytical predictions were within twelve percent of test results. The beam/frame web (shear) intersection static tests showed problem areas with bond strength to the woven graphite material and induced tensile (peel) stresses. Using a film adhesive, instead of relying on the matrix bond alone, resulted in meeting the design ultimate load with a small margin of safety. The projected weight reductions using these joint concepts and all composite structures are 26 percent for the frames and 30 percent for the transmission beams.

572. Krauss, T. A.: *Bearingless Helicopter Main Rotor Development. Volume 1: Design and Test Evaluation of Large Scale Composite Bearingless Main Rotor Flexbeam.* NASA CR-145188, June 1977.

77X10135#

An evaluation is presented of the critical structural components of a composite bearingless main rotor. The basic concept utilizes a flexbeam made of composite material to accommodate the flap, lag, and pitch motions of a rotor blade. The program is involved primarily with the design structural analysis, fabrication, and large scale structural testing of the critical flexbeam component which interfaces with the rotor shaft and rotor blades with simple bolted attachments.

573. Krauss, T. A.: *Advanced Systems Design Study of a Composite Structures Rotor for the RSRA. Volume 1: Design, Analysis, and Development Plan.* NASA CR-145082, Nov. 1976.

77X10008#

Studies were conducted to conceptually design and analyze a composite structures rotor (CSR) for the Rotor Systems Research Aircraft (RSRA). In addition a development plan was prepared outlining the steps required to deliver a flightworthy CSR to the government for flight evaluation.

574. Lowry, D. W.; and Rich, M. J.: *Design, Fabrication, Installation and Flight Service Evaluation of a Composite Cargo Ramp Skin on a Model CH-53 Helicopter.* NASA CR-172126, Apr. 1983.

83N33955#

The installation of a composite skin panel on the cargo ramp of a CH-53 marine helicopter is discussed. The composite material is Kevlar/Epoxy (K/E) which replaces aluminum outer skins on the aft two bays of the ramp. The cargo ramp aft region was selected as being a helicopter airframe surface subjected to possible significant field damage and would permit an evaluation of the long term durability of the composite skin panel.

575. Lowry, D. W.; Krebs, N. E.; and Dobyns, A. L.: *Design, Fabrication and Test of Composite Curved Frames for Helicopter Fuselage Structure.* NASA CR-172438, Oct. 1984.

85N25247#

Aspects of curved beam effects and their importance in designing composite frame structures are discussed. A detailed finite element analysis was conducted and used in the design of composite curved frame specimens. Five specimens were statically tested and compared with predicted and test strains.

576. Lowry, D. W.; Kay, B. F.; and Rich, M. J.: *Study To Investigate Design, Fabrication and Test of Low-Cost Concepts for Large Hybrid Composite Helicopter Fuselage.* NASA CR-159384, Feb. 1981.

81X10119#

The design of a cabin roof structure which consists of all graphite/epoxy structural elements bonded as an assembly without any mechanical fasteners is examined. Program results verify a projection of twenty-three percent weight saving from the current production metal baseline. The results of the static tests conducted to fracture on two specimens show that thin skin shear panels can be designed to buckle provided a method of preventing peeling of the skin from other structural members is used. The analytically predicted axial and shear strains correlate well with test results.

577. Rich, M. J.; and Lowry, D. W.: Composite Curved Frames for Helicopter Fuselage Structure. *Composite Structures* (Proceedings of the National Specialists' Meeting, Philadelphia, Pa., Mar. 23–25, 1983), American Helicopter Society, 1984.

85A21396

This paper presents the results of analysis and testing of composite curved frames. A major frame was selected from the UH-60 Black Hawk helicopter and designed as a composite structure. The curved beam effects were expected to increase flange axial stresses and induce transverse bending. A NASTRAN finite element analysis was conducted and the results were used in the design of composite curved frame specimens. Three specimens were fabricated and five static tests were conducted. The NASTRAN analysis and test results are compared for axial, transverse, and web strains. Results show the curved beam effects are closely predicted by a NASTRAN analysis and the effects increase with loading on the composite frames.

578. Rich, M. J.; and Lowry, D. W.: Design, Analysis and Test of Composite Curved Frames for Helicopter Fuselage Structure. *24th AIAA Structures, Structural Dynamics and Materials Conference* (Lake Tahoe, Nev., May 2–4), *Collection of Technical Papers, Part 1*, 1983, pp. 730–737. (Available as AIAA Paper 83-1005.)

83A29805#

Curved beam effects in composite frame structures representative of a light helicopter airframe are examined, and currently available analytical methods for studying these effects are briefly reviewed. A finite element study of curved composite frames is then reported, and it is shown that the curved frame effects must be accurately accounted for to avoid premature fracture. The finite element method is shown to be accurate within 10 percent in accounting for the curved beam effects in composite structures.

579. Rich, M. J.: Design, Fabrication and Test of a Complex Helicopter Airframe Section. SAE Paper 801213, 1980.

81A34221

A design solution is developed for the fabrication of an all-composite helicopter airframe cabin roof structure. The bonded graphite/epoxy elements of the structure employed aluminum tooling with control on all mating surfaces to yield accurate bond lines. A summary of static test results is presented.

580. Rich, M. J.; and Lowry, D. W.: *Flight Service Evaluation of Composite Helicopter Components.* NASA CR-172562, Apr. 1985.

85N34222#

An assessment of composite helicopter structures, exposed to environmental effects, after four years of commercial service is presented. This assessment is supported by test results of helicopter components and test panels which have been exposed to environmental effects since late 1979.

581. Valisetty, R. R.; and Rehfield, L. W.: *Simple Theoretical Models for Composite Rotor Blades.* NASA CR-175620, Nov. 1984.

85N22382

The development of theoretical rotor blade structural models for designs based upon composite construction is discussed. Care was exercised to include a number of nonclassical effects that previous experience indicated would be potentially important. A model, representative of the size of a main rotor blade, is analyzed in order to assess the importance of various influences. The findings of this model study suggest that a classical type theory is adequate. The potential of elastic tailoring is dramatically demonstrated.

582. Winter, R.; Pifko, A. B.; and Cronkhite, J. D.: Crash Simulation of Composite and Aluminum Helicopter Fuselages Using a Finite-Element Program. *20th AIAA Structures, Structural Dynamics and Materials Conference* (St. Louis, Mo., Apr. 4–6), *Technical Papers on Structures and Materials,* 1979, pp. 233–240. (Available as AIAA Paper 79-0781.)

79A29025#

This paper describes a mathematical investigation of the crash-impact responses of an all-composite helicopter cockpit section incorporating an energy absorbing concept and one of conventional aluminum construction, using the Grumman DYCAST finite-element nonlinear structural dynamics computer program.

583. Zinberg, H.: *Flight Service Evaluation of Composite Components on the Bell Helicopter Model 206L.* NASA CR-172296, Mar. 1984.

84N19479#

The flight service components for the Bell Model 206L JetRanger helicopter are examined. The components were placed in service in the Continental United States, Canada, and Alaska. The status of 34 sets of components is discussed. Approximately 27,500 flight hours were accumulated on the components as of August 1, 1983. Three sets of components and one-fifth of the exposure coupons were returned

and tested. The results are given. The overall behavior of the components and associated problems are discussed.

General aviation research

584. Adney, P. S.; and Horn, W. J.: *Design of a Composite Wing Extension for a General Aviation Aircraft.* NASA CR-173832, Sept. 1984.

84N32377#

A composite wing extension was designed for a typical general aviation aircraft to improve lift curve slope, dihedral effect, and lift to drag ratio. Advanced composite materials were used in the design to evaluate their use as primary structural components in general aviation aircraft. Extensive wind tunnel tests were used to evaluate six extension shapes. The extension shape chosen as the best choice was 28 inches long with a total area of 17 square feet. Subsequent flight tests showed the wing extension's predicted aerodynamic improvements to be correct. The structural design of the wing extension consisted of a hybrid laminate carbon core with outer layers of Kevlar—layed up over a foam interior which acted as an internal support.

585. Domack, C. S.; and Martin, G. L.: *Application of Selected Advanced Technologies to High Performance, Single-Engine, Business Airplanes.* NASA CR-172361, June 1984.

84N27719#

Improvements in performance and fuel efficiency are evaluated for five new configurations of a six place, single turboprop, business airplane derived from a conventional, aluminum construction baseline aircraft. Results show the greatest performance gains for enhancements in natural laminar flow. A conceptual diesel engine provides greater fuel efficiency but reduced performance. Less significant effects are produced by the utilization of composite materials construction or by reconfiguration from tractor to pusher propeller installation.

586. Dow, N. F.; Humphreys, E. A.; and Rosen, B. W.: *Guidelines for Composite Materials Research Related to General Aviation Aircraft.* NASA CR-3720, Sept. 1983.

84N11218#

Guidelines for research on composite materials directed toward the improvement of all aspects of their applicability for general aviation aircraft were developed from extensive studies of their performance, manufacturability, and cost effectiveness. Specific areas for research and for manufacturing development were identified and evaluated. Inputs developed from visits to manufacturers were used in part to guide these evaluations, particularly in the area of cost effectiveness. Throughout, the emphasis was to direct the research toward the requirements of general aviation aircraft, for which relatively low load intensities are encountered, economy of production is a prime requirement, and yet performance still commands a premium.

587. Kohlman, D. L.: Performance Improvements of Single-Engine Business Airplanes by the Integration of Advanced Technologies. DGLR Paper 82-064, 1982.

83A24178#

An assessment is presented of the performance gains and economic impact of the integration in general aviation aircraft of advanced technologies, relating to such aspects of design as propulsion, natural laminar flow, lift augmentation, unconventional configurations, and advanced aluminum and composite structures. All considerations are with reference to a baseline mission of 1300-nm range and 300-knot cruise speed with a 1300-lb payload, and a baseline aircraft with a 40-lb/sq ft wing loading and an aspect ratio of 8. Extensive analytical results are presented from the NASA-sponsored General Aviation Synthesis Program. Attention is given to the relative performance gains to be expected from the single-engined baseline aircraft's use of a low cost general aviation turbine engine, a spark-ignited reciprocating engine, a diesel engine, and a Wankel rotary engine.

588. Kohlman, D. L.; and Holmes, B. J.: Assessment of Advanced Technologies for High Performance Single-Engine Business Airplanes. *Proceedings of the 13th Congress of the International Council of the Aeronautical Sciences and AIAA Aircraft Systems and Technology Conference* (Seattle, Wash., Aug. 22-27), *Volume 1*, AIAA, 1982, pp. 512-563.

82A40932#

The prospects for significantly increasing the fuel efficiency and mission capability of single engine business aircraft through the incorporation of advanced propulsion, aerodynamics, and materials technologies are explored. It is found that turbine engines cannot match the fuel economy of the heavier rotary, diesel, and advanced spark reciprocating engines. The rotary engine yields the lightest and smallest aircraft for a given mission requirement and also offers greater simplicity and a multifuel capability. Great promise is also seen in the use of composite material primary structures in conjunction with laminar flow wing surfaces, a pusher propeller, and conventional wing-tail configuration. This study was conducted with the General Aviation Synthesis Program, which can furnish the most accurate mission performance calculations yet obtained.

589. Kohlman, D. L.; Matsuyama, G. T.; Hawley, K. E.; and Meredith, P. T.: *A Feasibility Study for Advanced Technology Integration for General Aviation.* NASA CR-159381, Nov. 1980.

81N15974#

An investigation was conducted to identify candidate technologies and specific developments which offer greatest promise for improving safety, fuel efficiency, performance, and utility of general aviation airplanes. Interviews were conducted with general aviation airframe and systems manufacturers and NASA research centers. The following technologies were evaluated for use in airplane design tradeoff studies conducted during the study: avionics, aerodynamics, configurations, structures, flight controls, and propulsion. Based on industry interviews and design tradeoff studies, several recommendations were made for further high payoff research. The most attractive technologies for use by the general aviation industry appear to be advanced engines, composite materials, natural laminar flow airfoils, and advanced integrated avionics systems. The integration of these technologies in airplane design can yield significant increases in speeds, ranges, and payloads over present aircraft.

Repair methods

590. Adkins, D. W.; and Pipes, R. B.: Planar Scarf Joints in Composite Repair. *ICCM/2* (Proceedings of the Second International Conference on Composite Materials, Toronto, Canada, Apr. 16-20), Metallurgical Society of AIME, 1978, pp. 1200-1210.

79A17048

This study examined the effects of varying certain important geometric parameters on the stresses in an adhesive repair of a thin composite laminate. The repair, a planar scarf joint with a doubler, was studied using finite element methods at scarf angles of 6 deg and 12 deg, damage lengths between 2.5 mm

and 25 mm, and doubler overlaps between 10 mm and 40 mm. Finite element models also showed that significant stress peaks occur at the ends of the scarf joint adhesive, similar to shear lag phenomena in lap joints. A bending model of the planar scarf joint with a doubler, based on mechanics of materials theory, was also developed.

591. Deaton, J. W.: *A Repair Technology Program at NASA on Composite Materials*. NASA TM-84505, Aug. 1982.

82N31447#

Repair techniques for graphite/epoxy and graphite/polyimide composite structures are discussed. Tension and compression test results for several basic repair processes that were applied to damaged specimens are shown to approach the strength of undamaged specimens. Other repair configurations currently under investigation are illustrated, and plans in the repair technology program are presented.

592. Deaton, J. W.; and Mosso, N. A.: Preliminary Evaluation of Large Area Bonding Processes for Repair of Graphite/Polyimide Composites. *Materials and Processes—Continuing Innovations* (Proceedings of the Twenty-Eighth National SAMPE Symposium and Exhibition, Anaheim, Calif., Apr. 12-14), Society for the Advancement of Material and Process Engineering, 1983, pp. 1425-1442.

84A17202

The present investigation is concerned with the results of research conducted to develop a bonding process applicable to the repair of large areas in Gr/PI composite structural components. Attention is given to five repair techniques for Gr/PI composite materials. The techniques were employed to fabricate large panels from which flexure and short-beam-shear specimens were machined.

593. Jones, J. S.; and Graves, S. R.: *Repair Techniques for Celion/LARC-160 Graphite/Polyimide Composite Structures*. NASA CR-3794, June 1984.

84N26752#

Research and development programs were initiated to develop repair processes and techniques specific to Celion/LARC-160 GR/PI structures with emphasis on highly loaded and lightly loaded compression critical structures for factory type repair. Repair processes include cocure and secondary bonding techniques applied under vacuum plus positive autoclave pressure. Viable repair designs and processes

are discussed for flat laminates, honeycomb sandwich panels, and hat-stiffened skin-stringer panels. The repair methodology was verified through structural element compression tests at room temperature and 315°C (600°F).

594. Knauss, J. F.; and Stone, R. H.: Demonstration of Repairability and Repair Quality on Graphite/Epoxy Structural Subelements. *Proceedings of the 27th National SAMPE Symposium and Exhibition* (San Diego, Calif., May 4-6), Society for the Advancement of Material and Process Engineering, 1982, pp. 874-885.

83A20485

The program objective is to develop and validate repair procedures for composite structures which are adaptable to commercial airline maintenance operations. Questions concerning the design and the fabrication of repairs are discussed, taking into account a vertical fin cover panel, a wing cover panel, and a vertical fin spar. The test results indicate that the use of graphite patch repairs, either precured bonded or employing cure-in-place, is satisfactory for the repair of lightly loaded and highly loaded parts. The use of bolted repairs is satisfactory for lightly loaded structural components. The repair of composite substructures can be accomplished using comparable approaches to those evaluated for skin cover repairs in many previous programs.

595. Pipes, R. B.; and Adkins, D. W.: *Strength and Mechanics of Bonded Scarf Joints for Repair of Composite Materials*. NASA CR-169708, Dec. 1982.

83N15359#

Experimental and analytical investigations of scarf joints indicate that slight bluntness of adherend tips induces adhesive stress concentrations which significantly reduce joint strength. The laminate stacking sequence can have important effects on the adhesive stress distribution. A significant improvement in joint strength is possible by increasing overlap at the expense of raising the repair slightly above the original surface.

596. Stein, B. A.; and Hodges, W. T.: Rapid Adhesive Bonding Concepts for Specimen and Panel Fabrication and Field Repair. *Proceedings of the 16th National Technical Conference* (Albuquerque, N. Mex., Oct. 9-11), Society for the Advancement of Material and Process Engineering, 1984, pp. 103-118.

85A37379

123

The NASA Langley Research Center has developed bonding concepts for aerospace composite materials which employ induction heating to directly apply heat to the bond line and/or adherends without simultaneously heating the entire structure, supports, and fixtures of a bonding assembly. These methods have demonstrated bonding process time reductions of two to three orders of magnitude, by comparison with conventional press molding. Attention is presently given to rapid adhesive bonding for lap shear specimens for aerospace panel bonding or field repair, as well as for the field repair requirements of metallic and advanced polymeric matrix composite structures.

597. Stone, R. H.: *Repair Techniques for Graphite/Epoxy Structures for Commercial Transport Applications.* NASA CR-159056, Jan. 1983.

83N16397#

Composite defect sensitivity and airline damage experience and repair capabilities were surveyed. Repair concepts were screened. Repair of subelement specimens is discussed.

598. Stone, R. H.: Development of Repair Procedures for Graphite/Epoxy Structures on Commercial Transports. *Materials 1980* (Proceedings of the Twelfth National Technical Conference, Seattle, Wash., Oct. 7-9), Society for the Advancement of Material and Process Engineering, 1980, pp. 688-701.

81A43645

The first phase of the program included a survey of airline damage experience and airline maintenance and repair capabilities for composite structures. A survey was also conducted for available data on composite damage tolerance. The second phase of the program evaluated various depot and field level repairs ranging from precured bonded graphite flush patches to mechanically attached aluminum patches. Based on airline survey results, the emphasis was on field repairs. The results verified the effectiveness of field type repairs for lightly loaded composite structure.

Space applications.

599. Bluck, R. M.; and Johnson, R. R.: *Fabrication of Slender Struts for Deployable Antennas.* NASA CR-172164, Apr. 1983.

83N31733#

A procedure for fabricating graphite/epoxy tubing with an aluminum foil inner and outer wrap was developed. The aluminum foil provides a vapor barrier, significantly improves the thermal conductivity, and provides an excellent thermal control surface.

600. Bluck, R. M.; and Johnson, R.: Low Cost Fabrication of Graphite Epoxy Column Elements for Large Space Structures. *The Enigma of the Eighties: Environment, Economics, Energy* (Proceedings of the Twenty-Fourth National Symposium and Exhibition, San Francisco, Calif., May 8-10), *Book 1*, Society for the Advancement of Material and Process Engineering, 1979, pp. 285-292.

79A43248

A procedure for fabricating graphite/epoxy column elements used in the construction of large space platforms is described. Dry fiber is wound on a tapered aluminum mandrel in the LMSC vertical winding machine, and resin is injected between the mandrel and an outer sleeve. The winding and injection take place at elevated temperature to minimize the thermal expansion problems that arise in curing a tube on an aluminum mandrel when the end fittings are integrally wound.

601. Bowles, D. E.; Tompkins, S. S.; and Sykes, G. F.: Electron Radiation Effects on the Thermal Expansion of Graphite/Resin Composites. AIAA Paper 84-1704, 1984.

84A37456#

The effects of 1 MeV electron radiation on the thermal expansion characteristics of two graphite reinforced resin matrix composite systems were studied. Specimens of both graphite/epoxy (T300/5208) and graphite/polyimide (C6000/PMR15) were irradiated. Dynamic mechanical analyses were performed to study changes in resin chemistry. Thermal expansion results indicate that radiation did produce permanent residual strains in the graphite/epoxy. However, no permanent changes in the coefficient of thermal expansion (CTE) were observed. No permanent residual strains or changes in the CTE attributable to radiation were observed for the graphite/polyimide specimens.

602. Bush, H. G.; and Mikulas, M. M., Jr.: *A Nestable Tapered Column Concept for Large Space Structures.* NASA TM X-73927, July 1976.

76N29358#

A structural element concept is described which permits achievement of weight critical payloads for space shuttle. These columns are highly efficient structural members which could be the basic building elements for very large, space truss structures. Parametric results are presented which show that untapered cylindrical columns result in volume limited payloads on the space shuttle and that nestable, tapered columns easily eliminate this problem.

603. Card, M. F.; and Jensen, J. K.: Preliminary Design of Tetrahedral Truss Reflector Structure. *The Microwave Radiometer Spacecraft*, NASA RP-1079, Dec. 1981, pp. 95–107.

82N16160#

A baseline structure proposed for the microwave radiometer spacecraft (MRS) reflector is a large graphite-epoxy truss. The truss structure was selected to provide adequate stiffness to minimize control problems and to provide a low-expansion "strong back" on which to mount and control reflector mesh panels. Details of the structural members, joints and assembly concepts are presented, a concept for the reflector mesh surface is discussed, and preliminary estimates of the mass and structural natural frequencies of the reflector system are presented.

604. Chang, C. K.; and Kamaratos, E.: *Theoretical Studies of Radiation Effects in Composite Materials for Space Use*. NASA CR-3618, Sept. 1982.

82N34335#

Tetraglycidyl 4,4'-diamino diphenyl methane epoxy cured with diamino diphenyl sulfone was used as a model compound. Computer programs were developed to calculate (1) energy deposition coefficients of protons and electrons of various energies at different depths of the material; (2) ranges of protons and electrons of various energies in the material; and (3) cumulative doses received by the composite in different geometric shapes placed in orbits of various altitudes and inclinations.

605. Cohan, H.; and Johnson, R. R.: A Unique Approach to Fabricating Precision Space Structures Elements. *Composite Structures* (Proceedings of the First International Conference, Paisley, Scotland, Sept. 16–18), Applied Science Publishers (London), 1981, pp. 580–591.

82A17533

A procedure of fabricating graphite/epoxy columns used in the assembly of large space platforms is described. The requirement for precise dimensional control led to a unique hot resin injection process. Dry, high modulus fiber is wound over a vertically mounted steam-heated mandrel. A steam-heated sleeve or caul is slipped over the wound mandrel and resin is injected and cured in place. Approximately 200 column elements have been fabricated using this efficient process.

606. Fogdall, L. B.; Cannaday, S. S.; and Slemp, W. S.: Effects of Space Radiation on Thin Polymers and Nonmetallics. AIAA Paper 77-741, 1977.

77A39510#

Advanced materials for various spacecraft systems in the 1980s and 1990s have been evaluated in situ after exposure to space radiation. Emphasis has been placed on materials having little or no previous base of environmental effects data. Applications ranging from earth orbit to near-sun have been covered. High temperature polymers and composites have been included.

607. Fornes, R. E.; Gilbert, R. D.; and Memory, J. D.: *Effects of High Energy Radiation on the Mechanical Properties of Epoxy-Graphite Fiber Reinforced Composites*. NASA CR-176487, Dec. 1985.

86N17476#

The changes in molecular structural and mechanical properties of epoxy-graphite fiber composites upon exposure to ionizing radiation in a simulated space environment were investigated following exposure to ionizing radiation. Cobalt-60 gamma radiation and 1/2 MeV electrons were used as radiation sources. The system was studied using electron spin resonance (ESR) spectroscopy, infrared absorption spectroscopy, contact angle measurements, and electron spectroscopy for chemical analysis.

608. Fornes, R. E.; Memory, J. D.; and Naranong, N.: *Effect of 1.33 MeV Gamma Radiation and 0.5 MeV Electrons on the Mechanical Properties of Graphite Fiber Composites*. NASA CR-169117, 1982.

82N27486#

Epoxy/graphite fiber, polyimide/graphite fiber, and polysulfone/graphite fiber composites were exposed to 1.33 MeV gamma irradiation and 0.5 MeV electron bombardment for varying periods of time. The effects of the irradiation treatments on the breaking stress and Young's modulus were studied

by a three point bending test. Effects were small; both electron radiation up to 5000 Mrad and gamma radiation up to 350 Mrad resulted in slight increases in both stress and modulus.

609. Giori, C.: Space Radiation Effects on Composite Matrix Materials—Analytical Approaches. *Proceedings of the Twenty-Fourth National Symposium and Exhibition,* San Francisco, Calif., May 8–10), *Book 2,* Society for the Advancement of Material and Process Engineering, 1979, pp. 1012–1020.

79A43305

In-vacuo ultraviolet and gamma radiation exposure tests were utilized in a study aimed at the identification of radiation damage mechanisms in composite materials, with the objective of predicting the long-term behavior of composite structures in a space environment at geosynchronous orbit. Physical and chemical methods of polymer characterization were utilized for the study of composite matrix degradation, in conjunction with GC/MS techniques for the analysis of volatile by-products.

610. Giori, C.; Yamauchi, T.; Rajan, K.; and Mell, R.: Mechanisms of Degradation of Graphite Composites in a Simulated Space Environment. AIAA Paper 83-0590, 1983.

83A16807#

Attention is given to degradation mechanisms for graphite/polysulfone and graphite/epoxy laminates exposed to ultraviolet and high-energy electron radiation in vacuum up to 960 equivalent sun hours and 10 to the 9th rads, respectively. The materials showed good electron radiation stability as indicated by the low G values for gas formation and no evidence of mechanical property changes. Quantum yields for gas formation indicate poor stability to ultraviolet radiation. Mechanical property measurements did not show significant changes up to 960 ESH, with the possible exception of P1700/C6000. The main products of irradiation were identified as hydrogen and methane, along with high levels of CO and CO_2.

611. Hagaman, J. A.: *A Survey of Structural Material Issues for a Space Station.* NASA TM-86385, Aug. 1985.

85N33168#

As NASA enters the definition phase of the space station project, one of the important issues to be considered is structural material selection. The complexity of the space station and its long life requirement are two key factors which must be considered

in the material selection process. Both aluminum and graphite/epoxy are considered as potential structural materials. Advantages and disadvantages of these materials with respect to mechanical and thermal considerations, space environment, manufacturing, and cost are discussed.

612. Heard, W. L., Jr.; Bush, H. G.; Wallsom, R. E.; and Jensen, J. K.: *A Mobile Work Station Concept for Mechanically Aided Astronaut Assembly of Large Space Trusses.* NASA TP-2108, Mar. 1983.

83N19806#

This report presents results of a series of truss assembly tests conducted to evaluate a mobile work station concept intended to mechanically assist astronaut manual assembly of erectable space trusses. The tests involved assembly of a tetrahedral truss beam by a pair of test subjects with and without pressure (space) suits, both in Earth gravity and in simulated zero gravity (neutral buoyancy in water). The beam was assembled from 38 identical graphite-epoxy nestable struts, 5.4 m in length with aluminum quick-attachment structural joints. Struts and joints were designed to closely simulate flight hardware.

613. Long, E. R., Jr.: *Electron and Proton Absorption Calculations for a Graphite/Epoxy Composite Model.* NASA TP-1568, Nov. 1979.

80N11226#

The Bethe-Bloch stopping power relations for inelastic collisions were used to determine the absorption of electron and proton energy in cured neat epoxy resin and the absorption of electron energy in a graphite/epoxy composite. Absorption of electron energy due to bremsstrahlung was determined. Electron energies for 0.2 to 4.0 MeV and proton energies from 0.3 to 1.75 MeV were used. Monoenergetic electron energy absorption profiles for models of pure graphite, cured neat epoxy resin, and graphite/epoxy composites are reported. A relation is determined for depth of uniform energy absorption in a composite as a function of fiber volume fraction and initial electron energy.

614. Mikulas, M. M., Jr.: *Structural Efficiency of Long Lightly Loaded Truss and Isogrid Columns for Space Applications.* NASA TM-78687, July 1978.

78N33480#

The general mass characteristics of long lightly loaded columns or space applications are investigated by studying four column concepts. The first is a simple tubular column, the second is a three longeron

truss column constructed of tubular members, the third is a three longeron truss column constructed of solid rod members, and the fourth is an open grid work isogrid wall tubular column. Design procedures, which include an initial imperfection in the straightness of the column, are developed for the different concepts and demonstrated numerically. A new set of structural efficiency parameters are developed for lightly loaded columns and are used to show a comparison of the masses of the four column concepts investigated.

615. Milkovich, S. M.; Herakovich, C. T.; and Sykes, G. F., Jr.: Space Radiation Effects on Graphite-Epoxy Composite Materials. NASA TM-85478, June 1984.

84N25769#

Radiation effects on engineering properties, dimensional stability, and chemistry of composite systems were characterized.

616. Penton, A. P.; Johnson, R., Jr.; and Freeman, V. L.: Fabrication of Composite Shell Structure for Advanced Space Transportation. *Selective Application of Materials for Products and Energy* (Proceedings of the Twenty-Third National Symposium and Exhibition, Anaheim, Calif., May 2–4), Society for the Advancement of Material and Process Engineering, 1978, pp. 137–149.

79A20810

For space structures that must resist buckling, graphite-epoxy materials offer an attractive potential for providing lightweight, low-cost structural components that will meet future space mission requirements. A description is presented of a program which was conducted to evaluate the merits of graphite-epoxy cylindrical shells and to continue the development of a design data base for ultralightweight structures. An objective of the program was to design, fabricate, and test a corrugated graphite-epoxy cylinder 10 ft in diameter and 10 ft long.

617. Phelps, H. R.; and Long, E. R., Jr.: Property Changes of Graphite/Epoxy Composite Exposed to Nonionizing Space Parameters. *Journal of Composite Materials*, vol. 14, Oct. 1980, pp. 334–341.

81A24498

A study has been made of the changes in the mass, thickness, and flexural properties of initially wet and dry specimens of graphite/epoxy composite

material due to the equivalent of eight weeks of exposure to nonionizing space environmental parameters. The parameters were near and middle solar UV irradiance, high vacuum, and temperature. The flexural properties were not affected by the exposures. Changes occurred to the mass, dimensions, and surface morphology of the specimens which varied with individual and combined parameter exposures. The combined UV and elevated thermal environment had synergistic effects on the properties of the specimens.

618. Rhodes, M. D.; and Mikulas, M. M., Jr.: *Composite Lattice Structure.* NASA TM X-72771, Sept. 1975.

75N32160#

A lattice type structural panel concept which exploits the unidirectional character of filamentary advance composite materials is described. This lattice has potential for application where stiff lightweight structures are needed such as large area panels for space satellites. Formulae are presented to calculate the panel weight and plate bending stiffness. This analysis indicates that structures with significantly lighter weight than conventional minimum gauge sandwich construction can be fabricated.

619. Sykes, G. F.; and Slemp, W. S.: Space Radiation Effects on an Elastomer-Toughened Epoxy-Graphite Composite. *Proceedings of the 30th National SAMPE Symposium and Exhibition* (Anaheim, Calif., Mar. 19–21), Society for the Advancement of Material and Process Engineering, 1985, pp. 1356–1368.

86A13167

An elastomer-toughened epoxy-graphite composite system (CE339/T300) was evaluated for its potential durability in the space radiation environment. The physical and chemical response of this system was characterized following exposure to radiation doses equivalent to 20 to 30 years in geosynchronous orbit using 1 MeV electrons. The results show that electrons generate extensive crosslinking embrittlement of the matrix. This embrittlement results in chemical and mechanical property changes that would limit the service life of this epoxy system in some space structure applications.

620. Tenney, D. R.; Sykes, G. F.; and Bowles, D. E.: Space Environmental Effects on Materials. Paper presented at the AGARD Specialists Meeting on Environmental Effects on Materials for Space Applications, Toronto, Canada, Sept. 22–24, 1982.

83A14125#

Research efforts at NASA Langley to characterize the durability of composite materials which are candidates for use as components on various space hardware systems are reviewed. The material applications include large space structures, antennas, cables, thermal control coating, solar reflectors, and satellite power systems.

Carbon fiber risk assessment

621. *Risk to the Public From Carbon Fibers Released in Civil Aircraft Accidents.* NASA SP-448, 1980.

81N18096#

As part of a Federal study of the potential hazard associated with the use of carbon fibers, NASA assessed the public risk associated with crash fire accidents of civil aircraft. The NASA study projected a dramatic increase in the use of carbon composites in civil aircraft and developed technical data to support the risk assessment. Personal injury was found to be extremely unlikely. In 1993, the year chosen as a focus for the study, the expected annual cost of damage caused by released carbon fibers is only $1000. Even the worst-case carbon fiber incident simulated (costing $178,000 once in 34,000 years) was relatively low-cost compared with the usual air transport accident cost. On the basis of these observations, the NASA study concluded that exploitation of composites should continue, that additional protection of avionics is unnecessary, and that development of alternate materials specifically to overcome this problem is not justified.

622. Alexander, J. G.: *Development of a Fire Test Facility for Graphite Fiber-Reinforced Composites.* NASA CR-159193, June 1980.

80N28443#

A number of modifications were added to a facility to make it specifically applicable for composite material screening tests. Most significant was the development of hardware for trapping fibers released during testing and isolating them for quantitative measurement. Capability was added for increasing test section velocities and increasing the range of air/fuel ratios available from very rich to very lean. A provision was added for agitation of the test specimen and the combustion gases by a pulsating gas supply.

623. Asad, N. N.: *Carbon/Graphite Fiber Risk Analysis and Assessment Study. Volume 2: Statistical Assessment of Fire Damage to Airframe Components.* NASA CR-159031, Apr. 1979.

79X10124#

The national risk associated with the accidental release of free carbon/graphite fibers from composite materials used in commercial aircraft is considered. Available jet transport aircraft accident data, domestic and foreign, are included. These data, in conjunction with the projected use of composite materials, can be used to predict the amount of carbon fiber released due to fire accidents at airports in the United States.

624. Babinsky, T. C.; and Musselman, K. A.: *Burn/Blast Tests of Miscellaneous Graphite Composite Parts.* NASA CR-163310, Nov. 1979.

80N24371#

This report describes the work performed by the Naval Surface Weapons Center on tests conducted for NASA Langley Research Center for a series of burn/blast tests with various graphite-reinforced composite materials and aircraft components. These tests were a continuation of a series requested by NASA, and, as in previous tests, information concerning the effects of fire and/or explosion upon aircraft structural composite materials and size distribution and dissemination patterns of released fiber materials for risk analyses was obtained.

625. Bell, V. L.: *The Potential for Damage From the Accidental Release of Conductive Carbon Fibers From Burning Composites.* NASA TM-80213, Apr. 1980.

80N18108#

The potential damage to electrical equipment caused by the release of carbon fibers from burning commercial airliners is assessed in terms of annual expected costs and maximum losses at low probabilities of occurrence. A materials research program to provide alternate or modified composite materials for aircraft structures is reviewed.

626. Bell, V. L.: The Potential for Damage From the Accidental Release of Conductive Carbon Fibers From Aircraft Composites. Paper presented at the AGARD Specialists Meeting on the Effect of Service Environment on Composite Materials, Athens, Greece, Apr. 16, 1980.

80A27596#

The paper considers the potential for damage from the accidental release of conductive carbon fibers from aircraft composites. The electrical conductivity of carbon and graphite fibers has led to

electrical equipment damage from the inadvertent release of virgin fibers into the atmosphere; an accidental release of carbon fibers from filamentary composites from the burning of crashed commercial airliners could damage electrical and electronic equipment. The experimental and analytical results by NASA of the methods of assessing the extent of potential damage in terms of costs are presented; the NASA materials research program to provide alternate or modified composites to overcome electrical hazards of carbon composites in aircraft structures is described.

627. Clarke, C. A.; and Brown, E. L.: *Assessment of Risk to Boeing Commercial Transport Aircraft From Carbon Fibers.* NASA CR-159211, June 1980.

81N17171#

The possible effects of free carbon fibers on aircraft avionic equipment operation, removal costs, and safety were investigated. Possible carbon fiber flow paths, flow rates, and transfer functions into the Boeing 707, 727, 737, and 747 aircraft and potentially vulnerable equipment were identified. Probabilities of equipment removal and probabilities of aircraft exposure to carbon fiber were derived.

628. Elber, W.: *A Probabilistic Analysis of Electrical Equipment Vulnerability to Carbon Fibers.* NASA TM-80217, Oct. 1980.

81N11113#

The statistical problems of airborne carbon fibers falling onto electrical circuits were idealized and analyzed. The probability of making contact between randomly oriented finite length fibers and sets of parallel conductors with various spacings and lengths was developed theoretically. The probability of multiple fibers joining to bridge a single gap between conductors or forming continuous networks is included. From these theoretical considerations, practical statistical analyses to assess the likelihood of causing electrical malfunctions were produced. The statistics obtained were confirmed by comparison with results of controlled experiments.

629. Galasso, F. S.; Scola, D. A.; and Veltri, R. D.: *Coatings for Graphite Fibers.* NASA CR-159304, Aug. 1980.

80N32439#

Graphite fibers released from composites during burning or an explosion caused shorting of electrical and electronic equipment. Silicon carbide, silica, silicon nitride, and boron nitride were coated on graphite fibers to increase their electrical resistances.

Resistances as high as three orders of magnitude higher than uncoated fiber were attained without any significant degradation of the substrate fiber. An organo-silicone approach to produce coated fibers with high electrical resistance was also used.

630. Kalelkar, A. S.; Fiksel, J.; Raj, P. P. K.; and Rosenfield, D. B.: *An Assessment of the Risks Presented by the Use of Carbon Fiber Composites in Commercial Aviation. Volume 1: Final Report. Volume 2: Supporting Appendices.* NASA CR-158989, Jan. 1979.

79N18074#

Carbon fiber (CF) composites are being used to an increasing extent in commercial aircraft, due to their excellent structural properties. Since carbon fibers are highly conductive, a potential risk was identified in the event that an aircraft with CF composite structures is involved in an accidental fire. If carbon fibers are released from the fire, they could disperse in the atmosphere and eventually cause damaging short circuits in electronic equipment at remote locations. This phenomenon could conceivably result in economic losses. The purpose of this study was to assess the risks presented to the nation as a whole by the use of CF composites in commercial aircraft, in terms of the potential economic losses from air carrier accidents.

631. Larocque, G. R.: *An Assessment of Power System Vulnerability To Release of Carbon Fibers During Commercial Aviation Accidents.* NASA CR-159208, Jan. 1980.

80N26394#

The vulnerability of a power distribution system in Bedford and Lexington, Massachusetts, to power outages as a result of exposure to carbon fibers released in a commercial aviation accident in 1993 was examined. Possible crash scenarios at Logan Airport based on current operational data and estimated carbon fiber usage levels were used to predict exposure levels and occurrence probabilities. The analysis predicts a mean time between carbon fiber induced power outages of 2300 years with an expected annual consequence of 0.7 persons losing power.

632. Lieberman, P.; Chovit, A. R.; Sussholz, B.; and Korman, H. F.: *Data Reduction and Analysis of Graphite Fiber Release Experiments.* NASA CR-159032, July 1979.

80N19048#

The burn and burn/explode effects on aircraft structures were examined in a series of fifteen outdoor tests conducted to verify the results obtained in previous burn and explode tests of carbon/graphite composite samples conducted in a closed chamber and to simulate aircraft accident scenarios in which carbon/graphite fibers would be released.

633. Paul, J. T., Jr.; and Weldy, W. E.: *Graphite Fiber Surface Treatment To Improve Char Retention and Increase Fiber Clumping.* NASA CR-159357, Nov. 1980.

81N13091#

Composites containing carbon and graphite fibers can release fibers into the atmosphere during a fire. This release can potentially cause failure in some types of electrical equipment. Reduced fiber dispersion during and after combustion will reduce risks. Epoxidized char forming systems were synthesized which will react with commercially available surface treated carbon fiber. Fibers modified with these char formers retained adhesion in a specific epoxy matrix resin. Small scale combustion testing indicates that using these char-former-modified fibers in laminates will help to reduce the dispersement of fibers resulting from exposure to fire without sacrificing resin to fiber adhesion.

634. Pocinki, L.; Cornell, M.; and Kaplan, L.: *Advanced Risk Assessment of the Effects of Graphite Fibers on Electronic and Electric Equipment.* NASA CR-159210, May 1980.

80N26393#

An assessment of the risk associated with accidents involving aircraft with carbon fiber composite structural components is examined. The individual fiber segments cause electrical and electronic equipment to fail under certain operating conditions. A Monte Carlo simulation model was used to compute the risk. Aircraft accidents with fire, release of carbon fiber material, entrainment of carbon fibers in a smoke plume, transport of fibers downwind, transfer of some fibers into the interior of buildings, failures of electrical and electronic equipment, and economic impact of failures are discussed. Risk profiles were prepared for individual airports and the Nation. The vulnerability of electrical transmission equipment to carbon fiber incursion and aircraft accident total costs are investigated.

635. Pride, R. A.: *Large-Scale Carbon Fiber Tests.* NASA TM-80218, Dec. 1980.

81N16137#

A realistic release of carbon fibers was established by burning a minimum of 45 kg of carbon fiber composite aircraft structural components in each of five large scale, outdoor aviation jet fuel fire tests. This release was quantified by several independent assessments with various instruments developed specifically for these tests. The most likely values for the mass of single carbon fibers released ranged from 0.2 percent of the initial mass of carbon fiber for the source tests (zero wind velocity) to a maximum of 0.6 percent of the initial carbon fiber mass for dissemination tests (5 to 6 m/s wind velocity). Footprints of downwind dissemination of the fire released fibers were measured to 19.1 km from the fire.

636. Pride, R. A.: *Carbon Fiber Counting.* NASA TM-80117, June 1980.

80N28446#

A method was developed for characterizing the number and lengths of carbon fibers accidentally released by the burning of composite portions of civil aircraft structure in a jet fuel fire after an accident. Representative samplings of carbon fibers collected on transparent sticky film were counted from photographic enlargements with a computer aided technique which also provided fiber lengths.

637. Pride, R. A.; McHatton, A. D.; and Musselman, K. A.: *Electronic Equipment Vulnerability to Fire Released Carbon Fibers.* NASA TM-80219, Sept. 1980.

80N33491#

The vulnerability of electronic equipment to damage by carbon fibers released from burning aircraft type structural composite materials was investigated. Tests were conducted on commercially available stereo power amplifiers which showed that the equipment was damaged by fire released carbon fibers but not by the composite resin residue, soot, and products of the combustion of the fuel associated with burning the carbon fiber composites. Results indicate that the failure rates of the equipment exposed to the fire released fibers were consistent with predictions based on tests using virgin fibers.

638. Rosenfield, D.; and Fiksel, J.: *An Assessment of the Risk Arising From Electrical Effects Associated With the Release of Carbon Fibers From General Aviation Aircraft Fires.* NASA CR-159206, Feb. 1980.

80N26391#

A Poisson type model was developed and exercised to estimate the risk of economic losses through 1993 due to potential electric effects of carbon fibers released from United States general aviation aircraft in the aftermath of a fire. Of the expected 354 annual general aviation aircraft accidents with fire projected for 1993, approximately 88 could involve carbon fibers. The average annual loss was estimated to be about $250 (1977 dollars) and the likelihood of exceeding $107,000 (1977 dollars) in annual loss in any one year was estimated to be at most one in ten thousand.

639. Rosenfield, D. B.; Fiksel, J.; and Kalelkar, A. S.: *An Assessment of the Risks Presented by Carbon Fiber Composites Released From Motor Vehicle Fires.* NASA CR-159207, Dec. 1979.

80N22408#

A risk assessment was conducted to estimate the potential losses through 1993 due to the usage of carbon fiber (CF) composites in U.S. motor vehicles, including automobiles and trucks. Motor vehicle fires could conceivably release minute carbon fibers which might disperse in the atmosphere, penetrate buildings or enclosures, and cause damaging shorts to electronic equipment. Of a total estimated 310,000 vehicle fires per year in the U.S., approximately 94,000 could potentially release carbon fibers. The expected loss was $5,567 per year (1977 dollars), and the likelihood of exceeding $500,000 in annual losses was estimated to be at most one in ten thousand.

640. Shepler, R. E.: *Modified Carbon Fibers To Improve Composite Properties.* NASA CR-159057, June 1979.

79N25146#

Thin coatings, 5 to 10 wt. percent, were applied to PAN-based carbon fibers. These coatings were intended to make the carbon fibers less electrically conductive or to cause fibers to stick together when a carbon fiber/epoxy composite burned. The effectiveness of the coatings in these regards was evaluated in burn tests with a test rig designed to simulate burning, impact, and wind conditions which might release carbon fibers.

641. Suplinskas, R. J.; and Henze, T. W.: *A Study of the Deposition of Carbide Coatings on Graphite Fibers.* NASA CR-159133, Sept. 1979.

79N33260#

The chemical vapor deposition of boron carbide and silicon carbide on graphite fibers to increase their electrical resistance was studied. Silicon carbide coatings were applied without degradation of the mechanical properties of the filaments. These coatings typically added 1000 ohms to the resistance of a filament as measured between two mercury pools. When SiC-coated filaments were oxidized by refluxing in boiling phosphoric acid, average resistance increased by an additional 1000 ohms; in addition resistance increases as high as 150 K ohms and breakdown voltages as high as 17 volts were noted. Data on boron carbide coatings indicated that such coatings would not be effective in increasing resistance and would degrade the mechanical properties.

642. Sussholz, B.: *Evaluation of Micron Size Carbon Fibers Released From Burning Graphite Composites.* NASA CR-159217, Apr. 1980.

80N21454#

Quantitative estimates were developed of micron carbon fibers released during the burning of graphite composites. Evidence was found of fibrillated particles which were the predominant source of the micron fiber data obtained from large pool fire tests. The fibrillation phenomena were attributed to fiber oxidation effects caused by the fire environment. Analysis of propane burn test records indicated that wind sources can cause considerable carbon fiber oxidation. Criteria estimates were determined for the number of micron carbon fibers released during an aircraft accident. An extreme case analysis indicated that the upper limit of the micron carbon fiber concentration level was only about half the permissible asbestos ceiling concentration level.

643. Tompkins, S. S.; and Brewer, W. D.: Effects of Boron and Glass Hybrid Epoxy/Composites on Graphite-Fiber Release in an Aircraft Fire. *The Enigma of the Eighties: Environment, Economics, Energy* (Proceedings of the Twenty-Fourth National Symposium and Exhibition, San Francisco, Calif., May 8–10), *Book 1*, Society for the Advancement of Material and Process Engineering, 1979, pp. 620–630.

79A43272

Graphite fibers are good electrical conductors and fibers released into the environment during a fire create a possible hazard to electrical equipment. Several graphite-epoxy hybrids were exposed to a fire and simulated explosion and their graphite fiber retention characteristics were examined. Several low melting-temperature glasses which wet and clump

graphite fibers and a glass/graphite fabric which reduced impact damage were identified as promising hybridizing components to minimize graphite fiber release.

644. Tompkins, S. S.; and Brewer, W. D.: *Preliminary Burn and Impact Tests of Hybrid Polymeric Composites.* NASA TM-78762, July 1978.

78N28177#

Free graphite fibers released into the environment from resin matrix composite components, as a result of fire and/or explosion, pose a potential hazard to electrical equipment. An approach to prevent the fibers from becoming airborne is to use hybrid composite materials which retain the fibers at the burn site. Test results are presented for three hybrid composites that were exposed to a simulation of an aircraft fire and explosion.

Langley summaries and overviews

645. Blankenship, C. P.; and Tenney, D. R.: Materials Technology for Large Space Structures. *Proceedings of the AFOSR Specialists Conference on Prime-Power for High Energy Space System, Volume 2,* R and D Associates, 1982.

83N15882#

Several of the key material technology needs that were identified for large space structures are outlined. They include lightweight structural materials, materials durability in the space environment, and some special aspects of materials fabrication technology. Examples of current materials research directed toward large space structures are described. Additional research needs and opportunities are noted. A short bibliography is included of selected references that describe large space structural concepts and related technology needs in detail.

646. Brooks, G. W.: Some Trends in Aircraft Design: Structures. *NASA/University Conference on Aeronautics,* NASA SP-372, 1975, pp. 213–272.

75N29014

Trends and programs currently underway on the national scene to improve the structural interface in the aircraft design process are discussed. The application of advanced fibrous composites, improved methods for structural analysis, and continued attention to important peripheral problems of aeroelastic and thermal stability are among the topics considered.

647. Card, M. F.: Trends in Aerospace Structures. *Astronautics and Aeronautics,* vol. 16, July–Aug. 1978, pp. 82–89.

78A43363#

Recent developments indicate that there may soon be a revolution in aerospace structures. Increases in allowable operational stress levels, utilization of high-strength, high-toughness materials, and new structural concepts will highlight this advancement. Improved titanium and aluminum alloys and high-modulus, high-strength advanced composites, with higher specific properties than aluminum and high-strength nickel alloys, are expected to be the principal materials. Significant advances in computer technology will cause major changes in the preliminary design cycle and permit solutions of otherwise too-complex interactive structural problems and thus the development of vehicles and components of higher performance. The energy crisis will have an impact on material costs and choices and will spur the development of more weight-efficient structures. There will also be significant spinoffs of aerospace structures technology, particularly in composites and design/analysis software.

648. Card, M. F.; and Rhodes, M. D.: Graphite-Epoxy Panel Compression Strength Reduction Due to Local Impact. Paper presented at the AGARD Specialists Meeting on the Effect of Service Environment on Composite Materials, Athens, Greece, Apr. 13–18, 1980.

80A27598#

A review of results from ongoing research to investigate the effects of low-velocity impact on the compressive strength of graphite-epoxy structures is presented. Extensive tests have been conducted on sandwich beams, laminated plates, and stiffened panels. Conditions for failures were investigated by impact tests on statically loaded test specimens. The effects of compression load intensity were such that lightly loaded graphite structures (such as aircraft secondary structures) were insensitive to impact damage. In more heavily loaded structures (such as wing panels), however, appreciable reductions in compressive strength occurred. The implications of the tests for structural design are discussed by comparing panel masses for designs where ultimate strains have been reduced due to impact considerations with the masses of designs with higher ultimate strains. Finally, preliminary test data is presented to show the possibility of improvements in damage tolerance achievable by using an alternate matrix material.

649. Heldenfels, R. R.: *Recent NASA Progress in Composites.* NASA TM X-72713, Aug. 1975.

75N29188#

The application of composites in aerospace vehicle structures is reviewed. Research and technology program results and specific applications to space vehicles, aircraft engines, and aircraft and helicopter structures are discussed in detail. Particular emphasis is given to flight service evaluation programs that are or will be accumulating substantial experience with secondary and primary structural components on military and commercial aircraft to increase confidence in their use.

650. Herring, R. N.; and Miller, D. H.: *Study of Potential for Sustained Supersonic Cruise Military Aircraft Utilizing Advanced Technologies: Optimized Fighter and Interceptor Concepts.* NASA CR-172337, May 1984.

84X10308#

Selected component and systems optimizations were conducted for three major areas of fighter aircraft design: aerodynamic configuration, propulsion system, and structural concepts and materials. An air defense study, including a mission analysis and sizing of an interceptor configuration, was conducted. These conceptual design activities extend a previous study of advanced supersonic cruise military aircraft. The resultant interceptor configuration has a takeoff gross weight of 58,700 lb to perform a Mach 2.0 intercept at a mission radius of 1000 nautical miles.

651. McComb, H. G., Jr.: Safe Structures for Future Aircraft. *Astronautics and Aeronautics,* vol. 21, Sept. 1983, pp. 72–75.

83A46350#

The failure mechanisms, design lessons, and test equipment employed by NASA in establishing the airworthiness and crashworthiness of aircraft components for commercial applications are described. The composites test programs have progressed to medium primary structures such as stabilizers and a vertical fin. The failures encountered to date have been due to the nonyielding nature of composites, which do not diffuse loads like metals, and the presence of eccentricities, irregular shapes, stiffness changes, and discontinuities that cause tension and shear.

652. McComb, H. G., Jr.: *Advanced Structures Technology and Aircraft Safety.* NASA TM-85664, July 1983.

83N29733#

NASA research and development on advanced aeronautical structures technology related to flight safety is reviewed. The effort is categorized as research in the technology base and projects sponsored by the Aircraft Energy Efficiency (ACEE) Project Office. Base technology research includes mechanics of composite structures, crash dynamics, and landing dynamics. The ACEE projects involve development and fabrication of selected composite structural components for existing commercial transport aircraft. Technology emanating from this research is intended to result in airframe structures with improved efficiency and safety.

653. Tenney, D. R.; and Dexter, H. B.: *Advances in Composites Technology.* NASA TM-86353, Jan. 1985.

85N17048#

Research is currently focused on the development of tough resins and high strain fibers in an effort to gain improved damage tolerance. Moderate success has been achieved with the development of new resins such as PEEK and additional improvements look promising with new thermoplastic resins. Development of innovative materials forms such as 2-D and 3-D woven fabrics and braided structural subelements is also expected to improve damage tolerance and durability of composites hardware. The new thrust in composites is to develop low cost manufacturing and design concepts to lower the cost of composite hardware. Processes being examined include automated material placement, filament winding, pultrusion, and thermoforming.

654. Tenney, D. R.; Sykes, G. F.; and Bowles, D. E.: Composite Materials for Space Structures. *Proceedings of 3rd European Symposium on Spacecraft Materials in Space Environment,* Nov. 1985, pp. 9–21.

86N22594#

The use of advanced composites for space structures is reviewed. Barriers likely to limit further applications of composites are discussed and highlights of research to improve composites are presented. Developments in composites technology which could impact spacecraft systems are reviewed to identify technology needs and opportunities.

Conference Documents

655. Douglas Aircraft Company: *ACEE Composite Structures Technology.* NASA CR-172359, Aug. 1984.

85X10222#

Progress in the development of technology for advanced composites in commercial aircraft is discussed. Commercial airframe manufacturers demonstrated technology readiness and cost effectiveness of advanced composites for secondary and medium primary components and initiated a concerted program to develop the data base required for efficient application of safety-of-flight wing and fuselage structure. Composites technology development programs are reviewed. Topics discussed include: damage tolerance and failsafe testing of the DC-10 composite vertical stabilizer; theory and analysis for optimization of composite multirow bolted joints; design and test of large wing joint demonstration components; and joints and cutouts in fuselage structure.

656. Boeing Commerical Airplane Company: *ACEE Composite Structures Technology.* NASA CR-172358, Aug. 1984.

85X10221#

The NASA Aircraft Energy Efficiency (ACEE) Composite Primary Aircraft Structures Program has made significant progress in the development of technology for advanced composites in commercial aircraft. Papers were presented on the following topics: (1) advanced composites on Boeing commercial airplanes; (2) composite wing panel durability and damage tolerance technology development; (3) design development of heavily loaded wing panels; and (4) pressure containment and damage tolerance in fuselage structure.

657. *Tough Composite Materials.* NASA CP-2334, May 1983.

85N12941#

Papers and working group summaries are presented which address composite material behavior and performance improvement. Topic areas include composite fracture toughness and impact characterization, constituent properties and interrelationships, and matrix synthesis and characterization.

658. *CTOL Transport Technology, 1978.* NASA CP-2036, Part 1, Feb. 1978.

78N27046#

Technology generated by NASA and specifically associated with advanced conventional takeoff and landing transport aircraft is reported. Topics covered include aircraft propulsion; structures and materials; and laminar flow control.

659. *Third Conference on Fibrous Composites in Flight Vehicle Design.* NASA TM X-3377, Part 1, Nov. 1975.

78N18133#

The use of fibrous composite materials in the design of aircraft and space vehicle structures and their impact on future vehicle systems are discussed. The topics covered include flight test work in composite components, design concepts and hardware, specialized applications, operational experience, and certification and design criteria. Contributions to the design technology base include data concerning material properties, design procedures, environmental exposure effects, manufacturing procedures, and flight service reliability. By including composites as baseline design materials, significant payoffs are expected in terms of reduced structural weight fractions, longer structural life, reduced fuel consumption, reduced structural complexity, and reduced manufacturing cost.

660. *Carbon Fiber Risk Analysis.* NASA CP-2074, Oct. 1978.

79N22196#

The scope and status of the effort to assess the risks associated with the accidental release of carbon/graphite fibers from civil aircraft are presented. Vulnerability of electrical and electronic equipment to carbon fibers, dispersal of carbon fibers, effectiveness of filtering systems, impact of fiber induced failures, and risk methodology are among the topics covered.

661. *Advances in Engineering Science, Volume 2.* NASA CP-2001, Vol. 2, Nov. 1976.

77N10265#

Papers are presented dealing with structural dynamics; structural synthesis; and the nonlinear analysis of structures, structural members, and composite structures and materials. Applications of mathematics and computer science are included.

662. *Advances in Engineering Science, Volume 1.* NASA CP-2001, Vol. 1, Nov. 1976.

77N10230#

Proceedings from a conference on engineering advances are presented, including materials science, fracture mechanics, and impact and vibration testing. The tensile strength and moisture transport of laminates are also discussed.

663. *ACEE Composite Structures Technology: Review of Selected NASA Research on Composite Materials and Structures.* NASA CP-2321, Aug. 1984.

84N29969#

The NASA Aircraft Energy Efficiency (ACEE) Program was designed to develop technology for advanced composites in commercial aircraft. Research on composite materials, aircraft structures, and aircraft design is presented herein. The following parameters of composite materials were addressed: residual strength, damage tolerance, toughness, tensile strength, impact resistance, buckling, and noise transmission within composite materials structures.

664. *ACEE Composite Structures Technology: Review of Selected DOD Programs.* NASA CP-2322, Aug. 1984.

84X10423#

The following topics were presented at a conference on the NASA Aircraft Energy Efficiency (ACEE) Composite Primary Aircraft Structures Program: manufacturing technology for large aircraft composite primary structures (fuselages) and composite wing structures, damage tolerance of composites, and improved design for military aircraft through the use of composite structures in the design.

665. *Selected NASA Research in Composite Materials and Structures.* NASA CP-2142, Aug. 1980.

80N28436#

Various aspects of the application of composite materials to aircraft structures are considered. Failure prediction techniques, buckling and postbuckling research, laminate fatigue analysis, damage tolerance, high temperature resin matrix composites, and electrical hazards of carbon fiber composites are among the topics discussed.

666. *Assessment of Carbon Fiber Electrical Effects.* NASA CP-2119, Dec. 1979.

80N19193#

The risks associated with the use of carbon fiber composites in civil aircraft are discussed along with the need for protection of civil aircraft equipment from fire-released carbon fibers. The size and number of carbon fibers released in civil aircraft crash fires, the downwind dissemination of the fibers, their penetration into buildings and equipment, and the vulnerability of electrical/electronic equipment to damage by the fibers are assessed.

667. Lockheed Corporation: *ACEE Composite Structures Technology.* NASA CR-172360, Aug. 1984.

85X10014#

Composites technology development programs are reviewed. Topics discussed include strength and hygeothermal response of L-1011 vertical fin components; composite wing fuel containment and damage tolerance; impact dynamics and acoustic transmission in fuselage structure; and transport wing technology development.

668. *Welding, Bonding, and Fastening—1984.* NASA CP-2387, Sept. 1985.

Papers were presented on technology developed in current research programs relevant to welding, bonding, and fastening of structural materials required in fabricating structures and mechanical systems used in the aerospace, hydrospace, and automotive industries. Topics covered in the conference included equipment, hardware, and materials used when welding, brazing, and soldering, mechanical fastening, explosive welding, use of unique selected joining techniques, adhesives bonding, and nondestructive evaluation. A concept of "The Factory of the Future" was presented, followed by advanced welding techniques, automated equipment for welding, welding in a cryogenic atmosphere, blind fastening, stress corrosion resistant fasteners, fastening equipment, explosive welding of different configurations and materials, solid-state bonding, electron beam welding, new adhesives, effects of cryogenics on adhesives, and new techniques and equipment for adhesive bonding.

Author Index

Report Documentation Page

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15. Supplementary Notes

16. Abstract

Composites research conducted at the Langley Research Center during the period from 1975 to 1986 is described, and an annotated bibliography of over 600 documents (with their abstracts) is presented. The research includes Langley basic technology and the composite primary structures element of the NASA Aircraft Energy Efficiency (ACEE) Program. The basic technology documents cited in the bibliography are grouped according to the research activity such as design and analysis, fatigue and fracture, and damage tolerance. The ACEE documents cover development of composite structures for transport aircraft.

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